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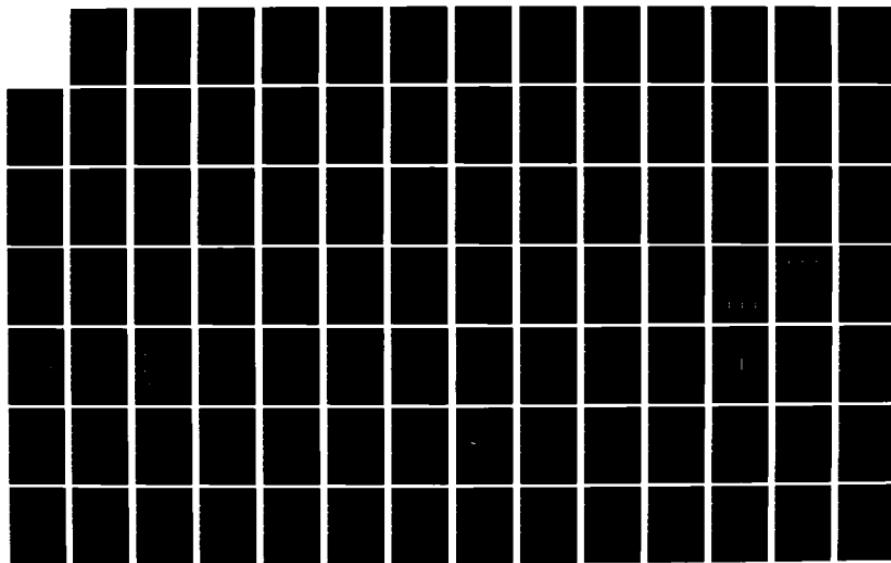
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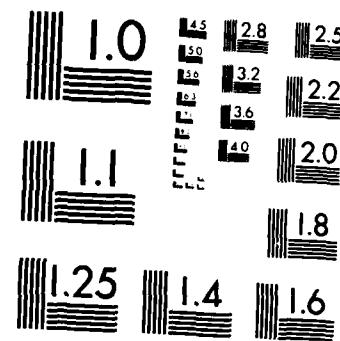
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W.W. Kaniuka et al

Aircraft and Crew Systems Technology Directorate
NAVAL AIR DEVELOPMENT CENTER
Warminster, Pennsylvania 18974

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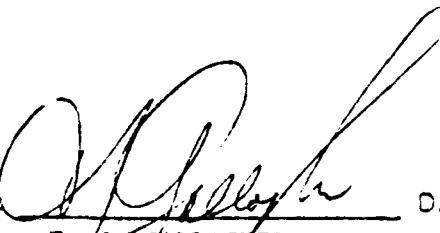
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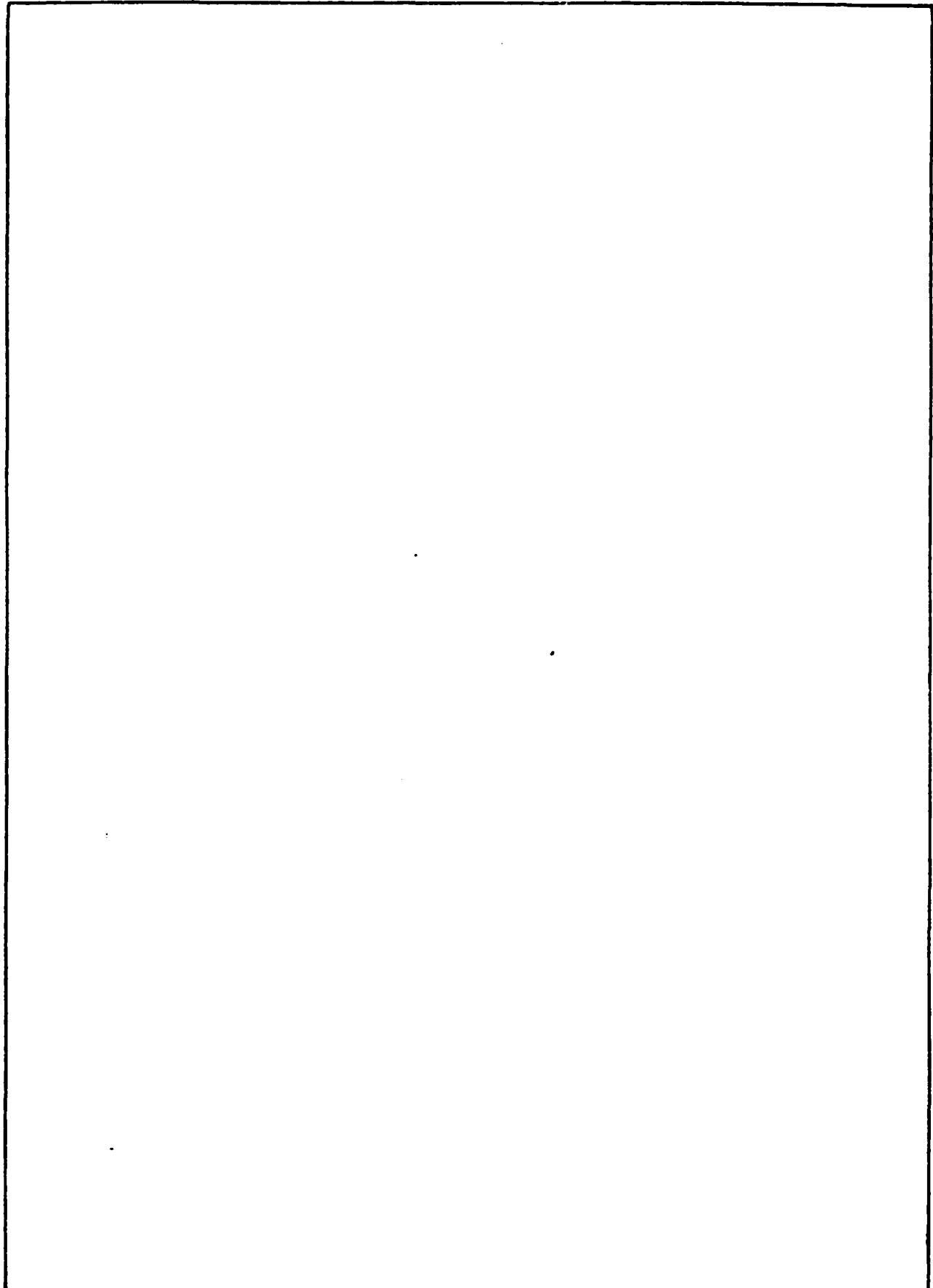
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20. ABSTRACT (Continue on reverse side if necessary and identify by block number) This report describes new flight control technologies that were developed in part or fully under Navy R&D funding for future advanced aircraft. The main thrust of these developmental programs was in the areas of digital flight control systems, advanced control surface actuators, and integrated sensory subsystems.		

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SUMMARY

High performance combat aircraft require Flight Control Systems (FCS) that permit the attainment of mission objectives within the constraints of cost, weight, volume, reliability and maintainability. The FCS must also be relatively invulnerable to enemy small projectile and radiation weaponry and natural hazards such as lightning.

The complexity and variety of tasks that FCS's are called upon to perform has been steadily increasing. However, the application of new technologies such as fly-by-wire, digital computation, and integrated systems will permit the design and synthesis of FCS's capable of meeting these highly demanding requirements.

- This report presents the new technologies and concepts whose development has been advanced by Navy supported Exploratory and Advanced Development Programs.

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LIST OF ABBREVIATIONS, ACRONYMS, AND SYMBOLS

AAG	Air-to-Air Gunnery
ABU	Analog Back Up
AC	Alternating Current
ACIA	Asynchronous Interface Adapter
A/D	Analog to Digital
ADS	Air Data System
AFCAS	Advanced Flight Control Actuation System
AFCS	Automatic Flight Control System
AFTI/F-16	Advanced Fighter Technology Integrator/F-16
AIU	Actuator Interface Unit
Δ N	Normal Acceleration
AOA	Angle of Attack
AOS	Angle of Sideslip
ARI	Aileron-Rudder-Interconnect
ASB	Air-to-Surface Bombing
ASG	Air-to-Surface Gunnery
ASSET	Advanced Skewed Sensory Electronic Triad
Δ X	Acceleration along the "X" Body Axis
Δ Y	Acceleration along the "Y" Body Axis
Δ Z	Acceleration along the "Z" Body Axis
BHT	Bell Helicopter Textron
BIT	Built-In-Test
$^{\circ}$ C	Degrees Celsius
CAD	Columbus Aircraft Division
CADC	Central Air Data Computer
CAS	Command Augmentation System
CCCM	Cross Channel Comparison Monitoring
CCSL	Coordinated Control-Surface Limiter
CCV	Control Configured Vehicle
CDS	Control Display System
C.G.	Center of Gravity
CIU	Control Interface Unit
cm	Centimeter
CM	Comparison Monitoring
CPU	Central Processor Unit
CVCM	Cross Voter Comparison Monitoring

LIST OF ABBREVIATIONS, ACRONYMS, AND SYMBOLS (Continued)

D/A	Digital to Analog
DAC	Digital to Analog Converter
DC	Direct Current
D ³	Direct Digital Drive
deg	Degree
DFBL	Digital Fly-By-Light
DFCS	Digital Flight Control System
DHS	Data Handling System
DLC	Direct Lift Control
DOF	Degree-of-Freedom
DP	Digital Processor
DSF	Direct Side Force
ECA	Electronic Component Assembly
EDU	Electronic Drive Unit
EHSV	Electro-Hydraulic Servo Valve
EMI	Electro-Magnetic Interference
EMP	Electro-Magnetic Pulse
EPROM	Erasable Programmable Read Only Memory
EPU	Emergency Power Unit
°F	Degrees Fahrenheit
FBW	Fly-By-Wire
FCC	Flight Control Computer
FCS	Flight Control System
FICR	Failure Isolation Computation Routine
FMDS	Failure Management and Display System
F/O	Fail-Operational
FTL	Fault Tolerance Level
g	Gravitational Constant
GAC	Grumman Aerospace Corporation
GD	General Dynamics
gm	Gram
gpm	Gallons per Minute
GSE	Ground Support Equipment
H.M.	Hinge Moment
HMS	Helmet Mounted Sight
HOFCAS	Hydra-Optic Flight Control Actuation System
HPA	Hydra-Powered Alternator

LIST OF ABBREVIATIONS, ACRONYMS, AND SYMBOLS (Continued)

Hp	Pressure Altitude Referenced to 29.92 in. Hg
H _{Pbc}	Pressure Altitude Referenced to Altimeter Barometric Setting
· Hp	Pressure Altitude Rate
Hr	Hour
H.T.	Horizontal Tail
HUD	Head-Up Display
Hz	Hertz (cycles per second)
IBU	Independent Back Up Unit
ICA	Inertial Component Assembly
IDC	Input Data Converter
IFBIT	In-Flight Built-In Test
IFFC	Integrated Fire and Flight Control
IFIM	In-Flight Integrity Management
INS	Inertial Navigation System
I/O	Input/Output
IOC	Input/Output Controller
ISA	Integrated Servoactuator
ISS	Integrated Sensory Subsystem
k	Kilo (10 ³)
kg	Kilogram
km	Kilometer
kVA	Kilo Volt Ampere
LCC	Life Cycle Costs
L/D	Lift to Drag Ratio
LEF	Leading Edge Flap
LHS	Lightweight Hydraulic System
L/m	Liters per Minute
LQS	Linear Quadratic Synthesis
LRU	Line Replaceable Unit
LSP	Low Speed Brake
LVDT	Linear Vuable Differential Transformer
M	Mach No.
MHD	Magnetic Heading Detector
MPa	Megapascals
MPD	Multi-Purpose Display
MPU	Microprocessor Unit
m/s	Meters per Second
msec	Millisecond
MUX	Multiplex Bus

LIST OF ABBREVIATIONS, ACRONYMS, AND SYMBOLS (Continued)

N	Newton (metric unit of force)
NAAD	North American Aircraft Division
NADC	Naval Air Development Center
ODC	Output Data Processor
OFP	Operational Flight Plan
p	Roll Rate
PDG	Programmable Display Generator
PDU	Power Drive Unit
PFCS	Primary Flight Control System
PIA	Peripheral Interface Adapter
PMG	Permanent Magnet Generator
PROM	Programmable Read Only Memory
Ps	Static Pressure
PSA	Pneumatic Sensor Assembly
psi	Pounds per Square Inch
P _T	Pitot Pressure
PVI	Pilot/Vehicle Interface
PWM	Pulse Width Modulation
P _α	AOA Sensing Port on Air Data Probe
P _β	AOS Sensing Port on Air Data Probe
q	Pitch Rate
q _c	Compressible Dynamic Pressure
r	Yaw Rate or Distance from Accelerometers to Virtual NAV-Base
RAM	Random Access Memory
RDMS	Redundancy Data Management System
ROM	Read Only Memory
RPM	Revolutions per Minute
RSS	Relaxed Static Stability
RVDT	Rotary Voltage Differential Transformer
s	Laplace Operator
SAHRS	Strapdown Attitude and Heading Reference System
SAS	Stability Augmentation System
SDOF	Single Degree of Freedom
sec	Second
SFCS	Secondary Flight Control System
SMS	Stores Management Set

LIST OF ABBREVIATIONS, ACRONYMS, AND SYMBOLS (Continued)

SSC	Side-Stick Controller
SV	Servo Valve
SVCR	Sensor Voting Computational Routine
SVSR	Sensor Voting Selection Routine
TEF	Trailing Edge Flap
TFRR	Transient Failure Removal Routine
TFAT	True Air Temperature (°F)
TM	Telemetry
TT	Total Temperature
VA	Volt Ampere
VC	Calibrated Airspeed
VDC	Volts Direct Current
VT	True Airspeed
VTOL	Vertical Take-Off Or Landing

Greek Symbols

α	Angle-of-Attack
α_q	Gyro Orientation Angle
α_1	Pitch Pointing Control Mode
α_2	Vertical Translation Control Mode
β	Angle-of-Sideslip
β	Angle-of-Sideslip Rate
β_a	Linear Accelerometer Central Half-angle
β_q	Gyro Central Half-angle
β_1	Yaw Pointing Control Mode
β_2	Lateral Translation Control Mode
δ	Deflection Angle
Δ	Differential
θ	Pitch Rate
μ_p	Microprocessor
σ	Air Density Ratio or One Standard Deviation
Φ	Roll Attitude
Φ	Roll Rate
Φ_a	Linear Accelerometer Orientation Angle
Ψ	Heading Attitude
ω	Angular Rate

INTRODUCTION

The Naval Air Development Center (NADC) has been and is currently sponsoring developmental programs to develop and/or apply advanced technologies applicable to the Flight Control Systems (FCS) of future tactical aircraft. The main thrust of these developmental programs is toward the development of an advanced type of FCS - a next generation Digital Flight Control System (DFCS) employing Fly-By-Wire (FBW) signal paths. The major goals of the developmental programs are as follows:

- Improvement of mission performance through flight control technology.
- Increased flight safety and mission reliability via redundancy data management.
- Reduced vulnerability to combat damage.
- Simplicity and reliability through standardized and interchangeable components.
- Improved maintainability through Built-In-Test (BIT) and diagnostics.
- Reduced weight and volume requirements.
- Immunity to electro-magnetic hazards/threats.

The flight control developmental programs sponsored by NADC and their associated contractors (as of report date) are listed as follows:

- Development of an Integrated Sensory Subsystem (ISS) - The Grumman Aerospace Corp.
- Development of a Digital Flight Control System for the Advanced Fighter Technology Integration AFTI/F-16 Aircraft - General Dynamics Corp., Fort Worth Division (A joint Navy, Air Force, and NASA program).
- Development of Two Advanced and Fault Tolerant Actuation Concepts for FBW Systems.
 - Direct Digital Drive (D³) High Pressure (8000 psi) Actuation Concept - Rockwell International Corp., Columbus Division.
 - Four-Valve Tandem FBW Actuator - Bell Helicopter Textron, Fort Worth.
- Development of the Hydra-Optic Flight Control Actuation System (HOFCAS) Concept - Rockwell International Corp., Columbus Division.

The ISS exploratory development program is now in its final phase and should be available for the advanced development phase in the near future.

The first phase of the AFTI/F-16 program, dedicated to the development of a triplex DFCS is still in progress and is scheduled to be completed in CY 1983. The first flight of the AFTI/F-16 took place in July 1982 and is now in its initial flight test program at the Edwards Air Force Base.

The six phases of the D³ program have been completed with the actuation concept being successfully test flown in a T-2C aircraft. A follow on program now in progress, is directed to the development of a compact FBW actuator with rotary type hydraulic servo valves, their electronic packages, and feedback paths imbedded within the cylindrical portion of the actuator body.

The first phase of the Four-Valve Tandem FBW Actuator program has been completed in July 1980. The concept has been successfully demonstrated via laboratory testing. The second phase of the program was directed toward the development of a FBW actuator with manual revision capability and was completed in November 1981.

The development of a laboratory version of the HOFCAS was completed in December 1980.

This report was prepared for the purpose of providing simplified descriptions of the concepts being developed. A follow-on report will be issued to document future flight control developmental programs and update existing programs.

DESCRIPTION OF THE INTEGRATED SENSORY SUBSYSTEM (ISS) CONCEPT

The Naval Air Development Center defined the requirement for an Integrated Sensory Subsystem (ISS) after observing the numerous sensory subsystems employed on their F-14 aircraft. Several subsystems on the aircraft had dedicated inertial and air data type sensors with the result being an inefficient and wasteful proliferation of sensors. Under contracts to the Naval Air Development Center (NADC) GAC evolved and developed the ISS concept which permits an optimal integration of sensors with redundant digital computers or microprocessors so that the outputs of a basic set of sensors can be shared and used by such aircraft subsystems as the following:

- Digital Flight Control System (DFCS).
- Cockpit instrumentation and displays.
- Engine inlet control.
- Engine thrust control.
- Wing sweep control.
- Inertial Navigation System (INS).
- Weapon delivery.

In addition, in order to meet very stringent reliability and survivability requirements future high performance combat aircraft will incorporate redundant systems and components, further complicating the sensor requirements.

A fully developed ISS will provide the following benefits:

- Increased flight safety and mission reliability for aircraft incorporated with Digital Fly-By-Wire (DFBW) systems and Relaxed Static Stability (RSS) through the use of redundant sensors and redundancy management.
- Decreased vulnerability to combat damage by the dispersal of components.
- Reduced parts count and inventory requirements.
- Reduced maintenance.
- Reduction of weight and volume.
- Increased operational readiness.

An ISS is composed of the following three main elements:

- (1) A functionally reliable and combat damage survivable sensor set consisting of the following:
 - Redundant low and high speed multi-purpose air data probes.
 - Six hard mounted, skewed, and dispersed rate integrating gyros.
 - Six hard mounted, skewed, and dispersed linear accelerometers.

- (2) An Input/Output (I/O) System that links the data from the redundant air and inertial data sensors to a redundant digital and/or distributed microprocessor complex for processing. The signal paths of the I/O system may be in the form of hard wiring, multiplex bus, or fiber optics, or any combination thereof.
- (3) A computation network within a redundant digital computer complex consisting of subroutines that provide:
 - Sensor data preprocessing to compensate for position errors and filtering of sensor outputs.
 - Sensor redundancy data management for sensor signal selection, failure detection, and failure isolation.
 - State estimation and sensor data normalization algorithms to account for deterministic errors associated with body bending coupling, dispersion of sensors, and random sensor noise.
 - Strapdown attitude and heading reference computations for the skewed and dispersed inertial sensors.
 - Air data computations for altitude, altitude rate, true airspeed, true Mach number, dynamic pressure, angle of attack (α), angle of sideslip (β), true static pressure, and total air temperature.

The calculated parameters resident in the computer memories are transmitted to the using subsystems on a time shared basis. Figure 1 is a Schematic and Signal Flow Diagram of the ISS Concept. The vectors shown in a conical arrangement represent the sensing axes of the six skewed rate gyros and the six skewed linear accelerometers. Inputs to the redundant flight control digital computer complex that are from systems not a part of the ISS are marked with an asterisk. The required computations for the ISS Concept, as illustrated in Figure 1, are performed in the redundant flight control digital computer complex on a time shared basis. All of the inputs to the computer complex are used in the flight control and navigation tasks. However, the inputs from the INS and the flux valves are also used to aid in the redundancy management of the ISS air data system.

Figure 2 is a schematic of the major components of the ISS and their associated signal flow paths. Each of the three Inertial Sensor Assemblies contain two skewed linear accelerometers and two skewed gyroscopes. The three multipurpose air data probes are able to sense pilot pressure (P_T), static pressure (P_S), and angle-of-attack (α). The nose mounted air data probe is also able to sense angle-of-sideslip (β). An additional method of sensing β is through sensing the difference in P_S on both sides of the fuselage. The two total temperature (T_T) probes, one on each side of the fuselage, are dualized to provide two fail-operational performance for this important air data parameter.

The ISS Inertial and Air Data Systems (ADS) provide two fail-operational performance. In the inertial sensor sets this performance level is obtained through the use of skewed gyroscopes, and skewed accelerometers (6 of each), and failure detection, isolation, and reconfiguration techniques. The ISS ADS provides the same two fail-operational performance level through failure detection, isolation, and analytical redundancy reconfiguration techniques. The methods of obtaining these two fail-operational performance levels will be described in detail later on.

ISS SENSOR SYSTEM CONFIGURATIONS

1. ISS Air Data System Sensors, Data Handling System, and Redundancy Data Management System

The core of the ISS Air Data System (ADS) is in its set of redundant multipurpose air data probes. Figure 3 shows a multipurpose air data probe configuration capable of sensing P_S , P_T , α , and β .

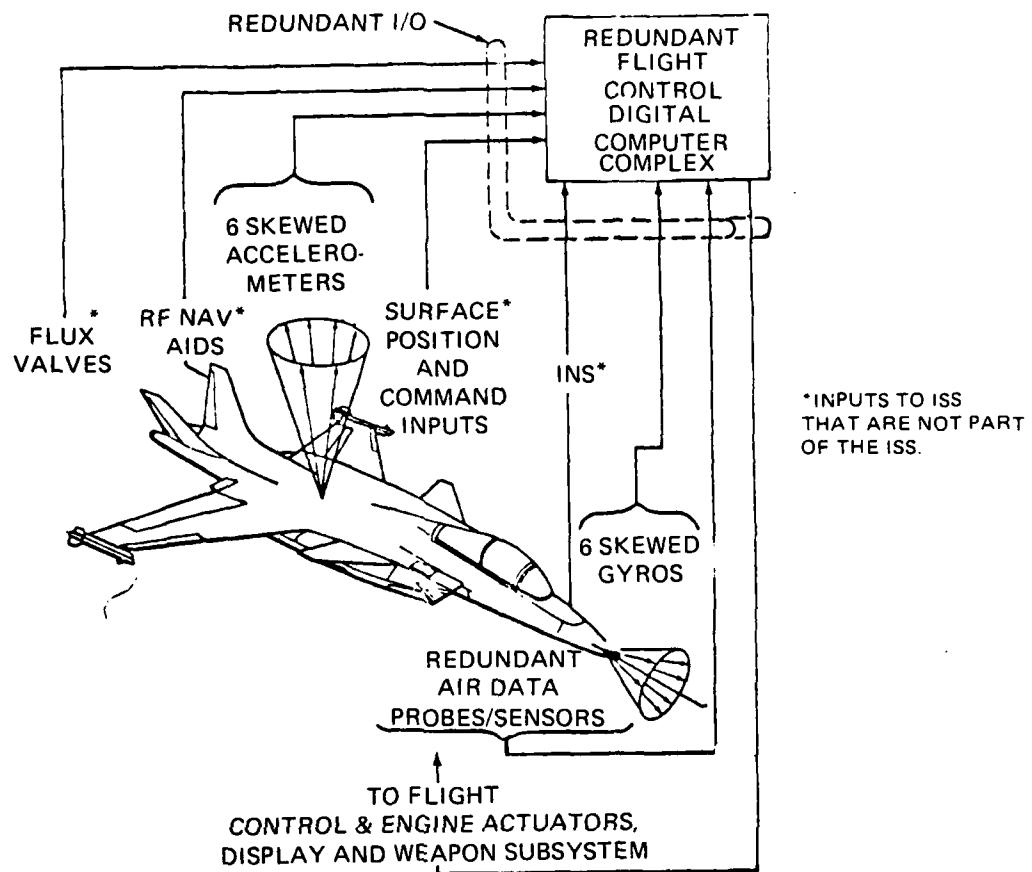


Figure 1. Schematic and Signal Flow Diagram of the ISS Concept

However, because of its hemispherical nose shape, the pitot pressure (P_T) indications are a function of α and β . With accurate position error corrections within the digital computer complex, true indications of P_T can be obtained.

Figure 4 shows the hemispherical sensor head of the multipurpose air data probe with two parts, labeled $P_{\alpha 1}$, and $P_{\alpha 2}$. With the aircraft at α equal to zero, the two parts sense equal pressures. At positive α , the pressure at the lower part ($P_{\alpha 1}$) becomes greater than the pressure at the upper port ($P_{\alpha 2}$). The greater the value of α in reference to the relative air flow, the greater the pressure differential between the two ports. α can be computed as a function of $(P_{\alpha 1} - P_{\alpha 2})$ for either positive or negative angles. However, this pressure differential increases with increasing dynamic pressure. Therefore, a normalizing function must be introduced to account for changes in dynamic pressure which is a function of altitude and airspeed. Normalization techniques for α and β measurements are described in Reference 1.

Additional probe errors are caused by local pressure disturbances and pressure variations at each probe location, which are functions of airspeed, Mach no., α , β , and the variations in flow geometry caused by flaps, spoilers, landing gear, and control surfaces.

The pressure and differential pressure outputs of the air data probes are sent to pressure transducers, as shown in Figure 2, via short pneumatic tubing runs to minimize the effect of transport lag. The

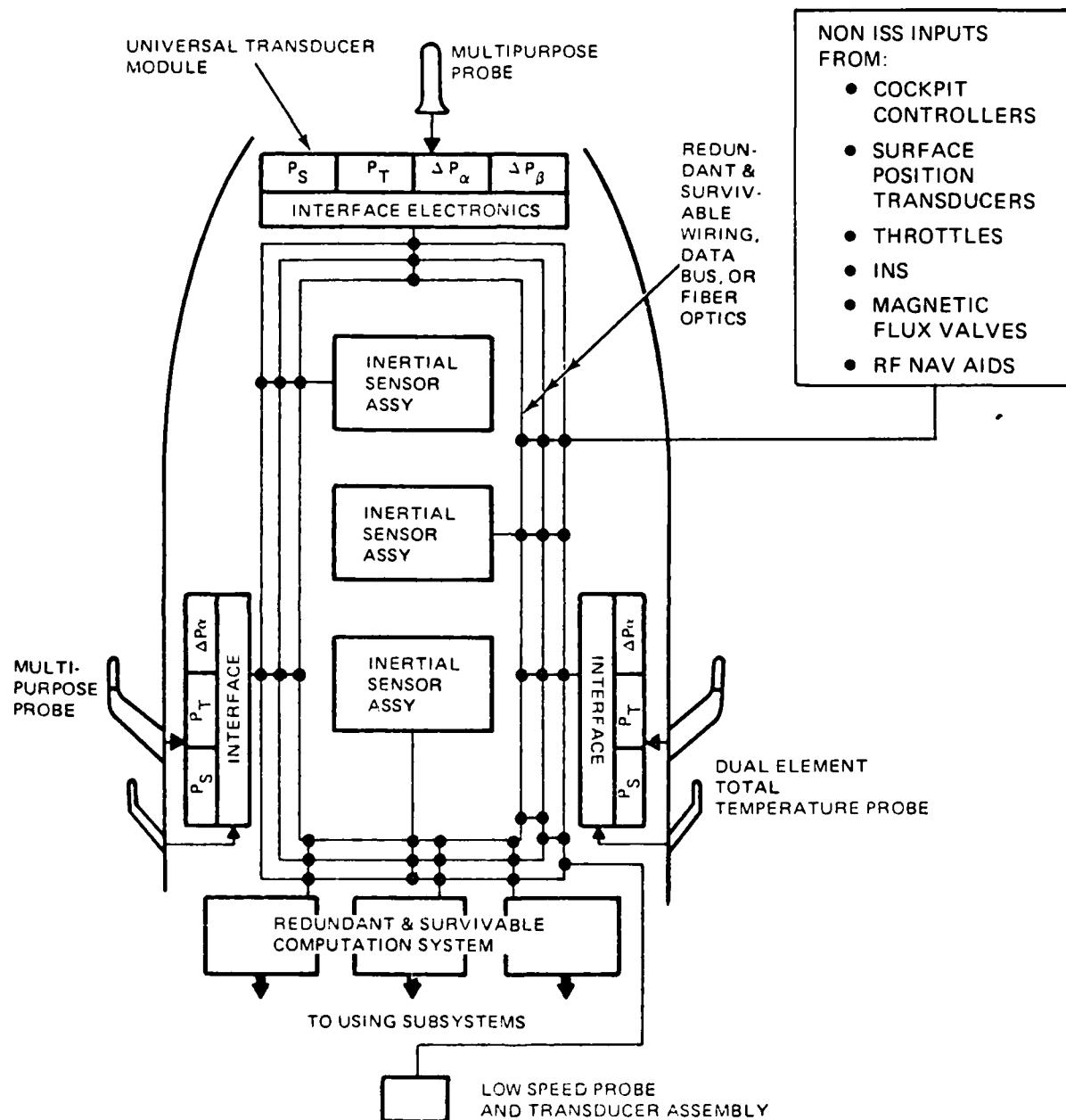


Figure 2. Schematic of the ISS Components and Associated Signal Flow Paths

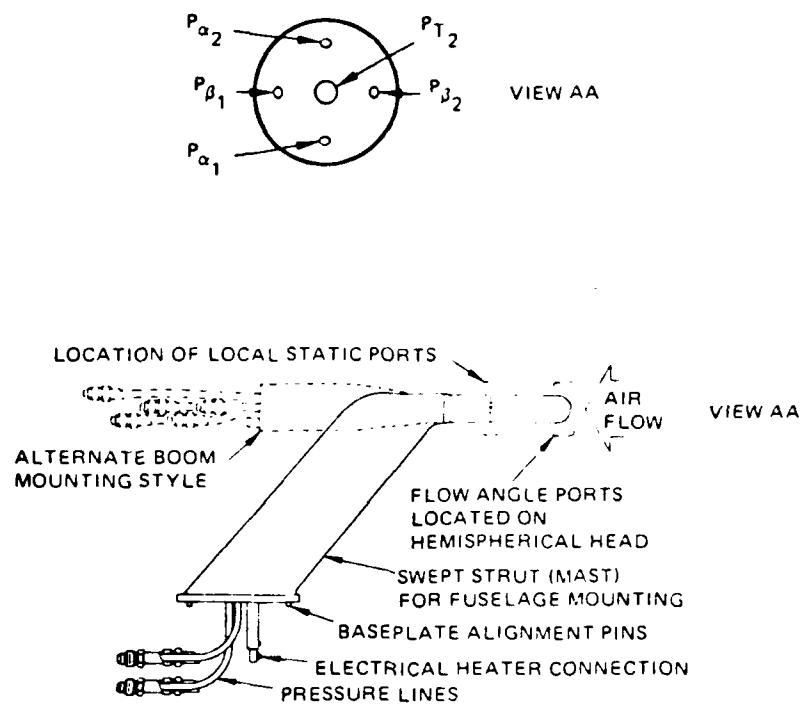
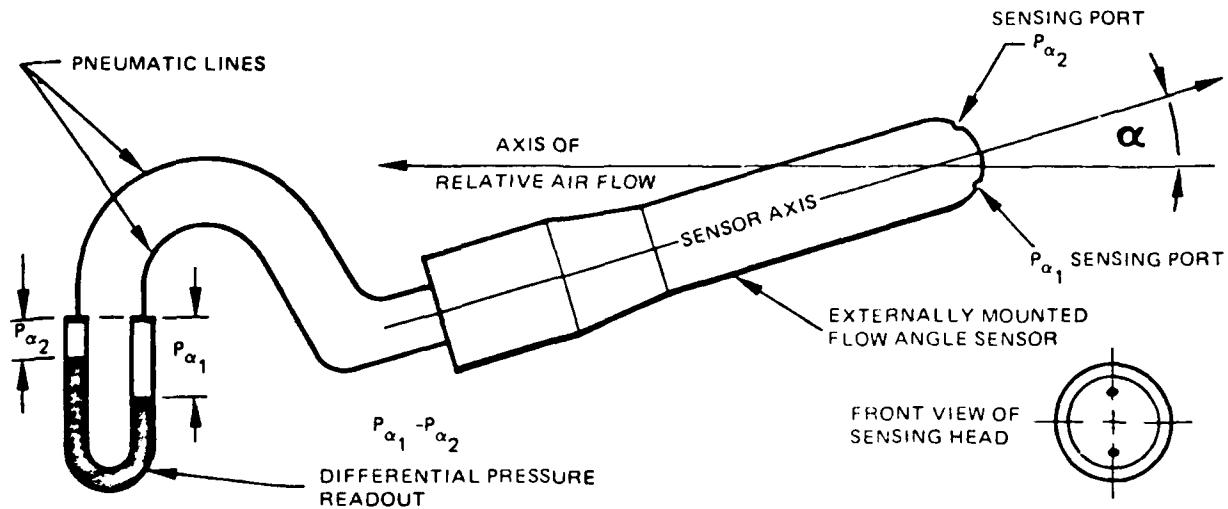


Figure 3. Multipurpose Air Data Probe

Figure 4. Pressure Differential with α

pressure transducers are to be mounted in a Universal Transducer Module (still in the development stage) which in addition to pressure transducers, will contain a microprocessor for preprocessing data and provide data output control, a power supply, and I/O electronics. The pressure transducer outputs are usually non-linear with respect to ambient pressure and input pressure. In order to correct for these non-linearities, the microprocessor will compute the required corrections based upon a predetermined set of linearization coefficients contained within each pressure transducer. After linearization, the corrected transducer data, under microprocessor control, is transmitted by the I/O electronics to the redundant computer complex via redundant transmission lines as shown in Figure 2. However, the linearized air data parameters transmitted to the redundant computer complex contain probe errors due to local pressure disturbances and local pressure variations. The differential pressures for α and β must also be corrected as functions of air speed and altitude.

In order to show the nature of the calculations required to remove these probe errors a signal flow diagram for a typical analog Central Air Data Computer (CADC) is shown in Figure 5. The diagram was obtained from Reference 2.

A Data Handling System (DHS) is integrated within the ISS Air Data System to perform the following functions:

- Normalization of indicated air data parameters.
- Redundancy data management.
- Air data computations.

Normalization is a multi-parameter compensation scheme which converts the linearized pressure transducer parameters from each probe location into useable air data parameters to perform the Redundancy Data Management System (RDMS) functions and the subsequent air data calculations to compute airspeed, altitude, altitude rate, Mach no., α , β , and total temperature. The local flow characteristics for the particular aircraft are required in order to accurately normalize the linearized pressure data. The RDMS within the DHS performs failure monitoring, failure isolation, and signal selection. In addition, through the use of analytical redundancy, a fourth channel of pressure ratio, α , and β are derived from a set of complementary filters, using inertial navigation data and valid air data parameters. Figure 6 outlines the methods employed by the RDMS to provide a two fail-operational level of performance. The outline was obtained from Reference 2. Figure 7 is a block diagram of the DHS. Figure 8 is a block diagram of the RDMS which is integrated within the DHS.

The failure monitoring that will be performed by the RDMS consists of two types as shown in Figure 8. The first type consists of a reasonableness test in which the latest calculated parameter is compared with the previously calculated parameter. If the parameter exceeds the failure threshold a transient failure is declared, and the associated data is not utilized in the subsequent calculations. The failure threshold is a function of aircraft dynamics and short term noise characteristics of the calculated parameter. For example, the threshold associated with the static pressure data will be a function of the maximum change in altitude of the aircraft, plus the short term noise characteristics of static pressure data that could occur between successive computation cycles.

The second type of failure monitoring performs comparisons between channels based on the most current data. The associated failure monitor thresholds will be functions of the following:

- Probe-to-probe reproducibility.
- Accuracy and repeatability of position error calibration.
- Air data transducer accuracy.

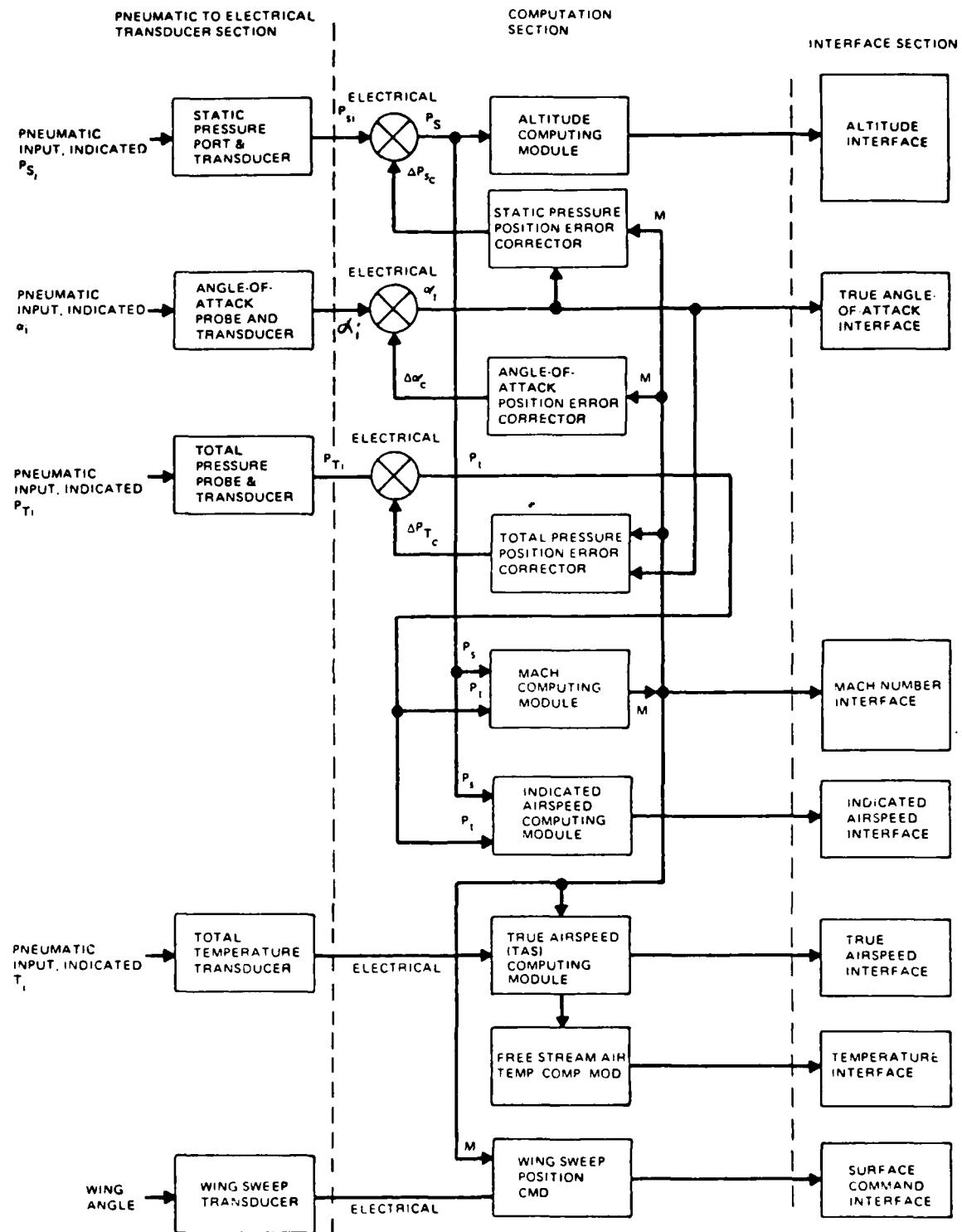
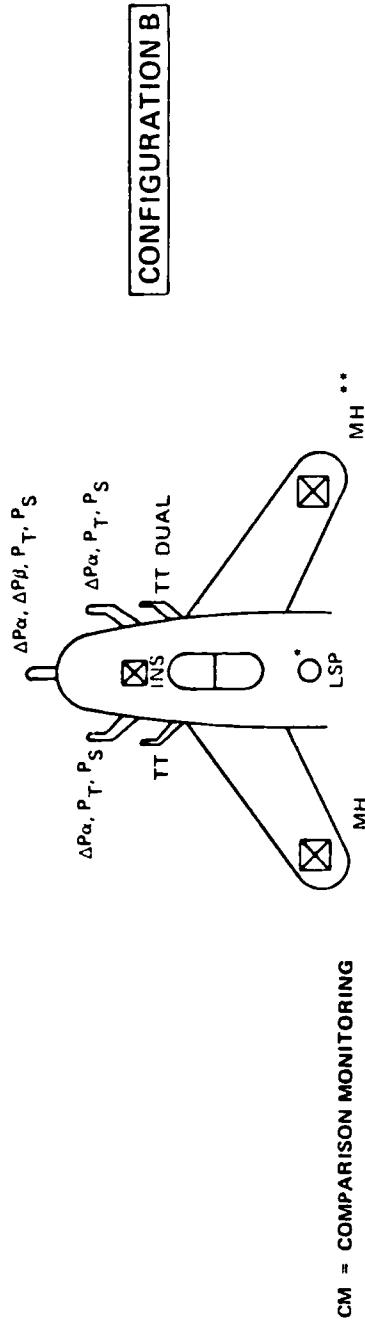


Figure 5. Computational Signal Flow in a Typical Analog Central Air Data Computer (CADC)



INFORMATION	1ST FAIL DETECTION	1ST FAILURE ISOLATION	SIGNAL SOURCE AFTER 1ST FAILURE	2ND FAIL DETECTION	2ND FAILURE ISOLATION	SIGNAL SOURCE AFTER 2ND FAILURE
STATIC PRESSURE (P_S)	CM	CM	2 REMAINING P_S	CM	CALCULATE TAS USING INS VELOCITY $\alpha, \beta, \theta, \psi, \phi$ AND SAVED WIND FROM PREVIOUS ITERATIONS. COMPARE TAS CALCULATED TO TAS DETERMINED WITH EACH OF DISCREPANT P_S OUTPUTS	REMAINING P_S
TOTAL PRESSURE (P_T)	CM	CM	2 REMAINING P_T	CM	CALCULATE TAS USING INS VELOCITY $\alpha, \beta, \theta, \psi, \phi$ AND SAVED WIND. COMPARE TAS CALCULATED TO TAS DETERMINED WITH EACH OF DISCREPANT P_T OUTPUTS	REMAINING P_T
ΔP_a	CM	CM	2 REMAINING ΔP_a	CM	CALCULATE α USING P_S , INS VELOCITY θ, ψ, ϕ , SAVED WIND. COMPARE TO REMAINING ΔP_a 's	REMAINING ΔP_a
TOTAL TEMPERATURE TT	CM	CM	2 REMAINING TT	CM	CALCULATE TAS USING INS VELOCITY $\alpha, \beta, \theta, \psi, \phi$ AND SAVED WIND. COMPARE TAS CALCULATED TO TAS DETERMINED WITH EACH DISCREPANT TT OUTPUT	REMAINING TT

- LOW SPEED SENSOR & TRANSDUCER ASSY
- .. MAGNETIC AZIMUTH DETECTOR

Figure 6. Outline of Air Data Redundancy Management System (RDMS) Functions (Sheet 1 of 2)

INFORMATION	1ST FAIL DETECTION	1ST FAILURE ISOLATION	SIGNAL SOURCE AFTER 1ST FAILURE	2ND FAIL DETECTION	2ND FAILURE ISOLATION	SIGNAL SOURCE AFTER 2ND FAILURE
ΔP_β TWO SOURCES: • NOSE PROBE (β_1) • Δ STATIC (β_2)	CM	IF STATIC SOURCES NOT FAILED AS-SUME β_1 FAILED	REMAINING ΔP_β	NOSE PROBE AVAILABLE (β_1) USE ΔP_α FOR DETECTION BECAUSE COMPUTATION OF α DEPENDS ON β OPENINGS A STATIC IN USE (β_2) USE STATIC SOURCE CM FOR DETECTION AND ISOLATION		• ONE SIDE β AVAILABLE OR ESTIMATE β TRIM USING ESTIMATOR (REF. 3)
HEADING 1.INS 1-SAHRs 2-MH	CM	CM	REMAINING HEADING SENSORS	CM	WHEN DISCREPANCY OF 2MH OR 1MH & INS OCCURS UPDATE TO SAHRs STOPS AND SAVED SAHRs HEADING IS USED FOR FAILURE ISOLATION	REMAINING HEADING SENSORS
PROBE FAILURE $P_\alpha, P_\beta, P_T, P_S$	CM	CM	REMAINING PROBE	CM	CALCULATE TAS WITH INS VELOCITY & SAVED WIND. COMPARE TO TAS DETERMINED WITH PROBES 1 & 2	REMAINING PROBE
LOW SPEED PROBE AND TRANSDUCER ASSY	COMPARISON OF PARAMETERS AVAILABLE FROM MULTIPURPOSE PROBE SET AND LSP IN REGION BETWEEN 200 KTS AND 60 KTS	SAME AS DETECTION	USE 'SUBSTITUTE' DESCRIBED IN FIGURE 37 OF REF. 2			

Figure 6. Outline of Air Data Redundancy Management System (RDMS) Functions (Sheet 2 of 2)

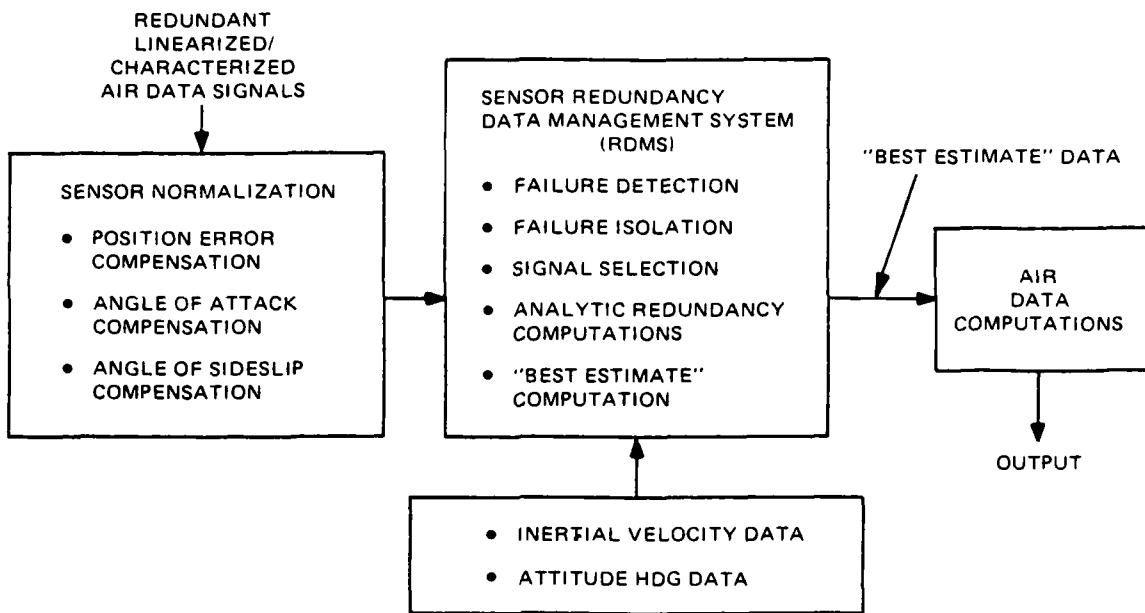


Figure 7. Block Diagram of the Data Handling System (DHS)

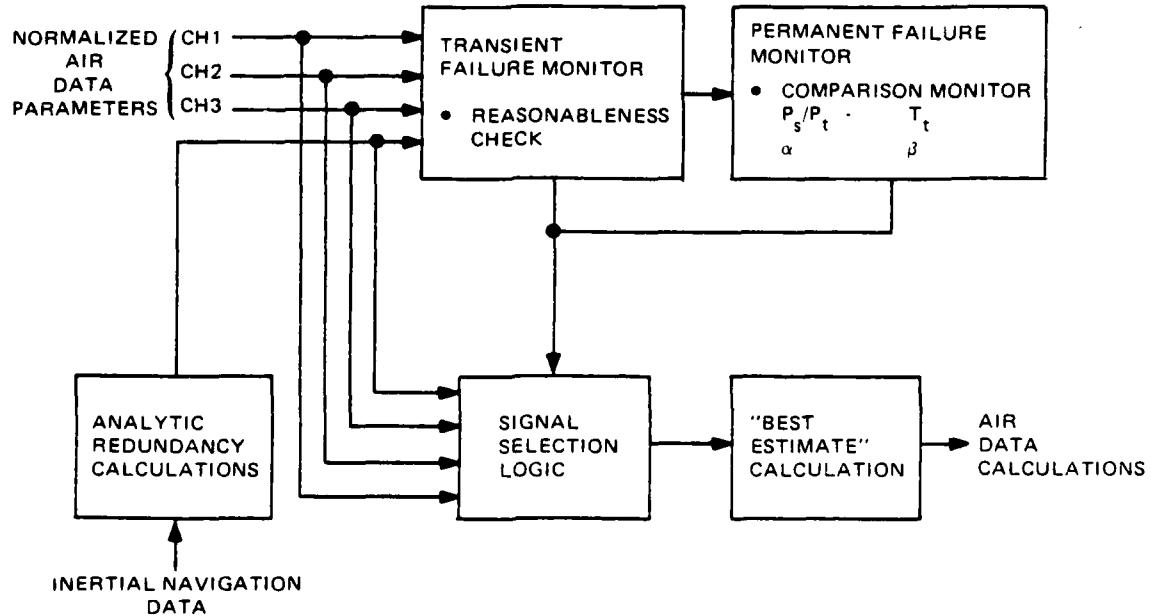


Figure 8. Block Diagram of the Redundancy Data Management System (RDMS)

- System dynamic response errors.
- Errors associated with analytical redundancy computations.

Should a parameter exceed the failure threshold for a specified length of time, a permanent failure is declared.

The signal selection routines in the DHS control the sensory data which will be utilized to determine the "best estimate" for each air data parameter. Should a transient or permanent failure occur, the associated sensory data is not utilized in the "best estimate" computations.

The final element in the DHS is the air data computations which utilize the "best estimate" sensory data from which true airspeed, Mach no., altitude, α , β , and etc. are calculated and then made available to the using subsystems via the I/O System of the ISS.

Table 1 shows how the air data parameters are shared among the using subsystems.

2. ISS Inertial Data System Sensors, Data Handling System, and Redundancy Data Management System

The ISS Inertial Data System uses six skewed rate gyros and six skewed linear accelerometers in order to obtain a two fail-operational performance level while minimizing the number of sensors and sensor types required. The six linear accelerometers have the same linear acceleration sensing range and the six gyros have the same angular rate sensing range. This feature minimizes the number of sensor types required. The refinement of the skewed sensor concept was advanced by the Grumman Aerospace Corporation during its development of the Advanced Skewed Sensory Electronic Triad (ASSET) System for Flight Control (References 4 and 5). The ASSET concept was successfully flight tested in an EA-6B aircraft.

The ASSET program was limited to the development of skewed rate gyro configurations. No effort was expended in the development of skewed linear accelerometer configurations. The sensing axis of each of the six rate gyros were skewed in a manner to permit sensing of components of pitch, roll, and yaw rates about the aircraft's body axes. This feature permitted the development of rate gyro configurations that provided dual fail-operational capability with only six rate gyros. In the

Table 1. Air Data Parameter Requirements Summary

PARAMETER	ALTITUDE	ALTITUDE RATE	AIR SPEED	MACH	DYNAMIC PRESSURE	ANGLE OF ATTACK	ANGLE OF SIDESLIP
SUBSYSTEM							
COCKPIT DISPLAYS	X	X	X	X		X	X
OUTER LOOP FLIGHT CONTROLS	X	X	X*	X		X	X
INNER LOOP FLIGHT CONTROLS				X	X	X	X
VARIABLE GEOMETRY INCLUDING AIR INLET & V/STOL ENGINE ORIENTATION	X		X	X	X	X	X
NAVIGATION WEAPONS SYSTEM	X	X	X	X		X	

*AS BACKUP TO INERTIAL VELOCITY AT LOW SPEEDS.

conventional approach of employing four rate gyros per body axis, mounted orthogonally, twelve rate gyros are required to obtain dual fail-operational capability (via use of majority voting techniques). Thus the ASSET approach permits the elimination of six rate gyros. Figure 9 illustrates the salient features of both the conventional and the ASSET concepts.

Figure 10 shows the geometry of the sensing or input axes of the six rate gyros which are oriented to position them on the surface of a cone, with the cone axis oriented along the aircraft's Z body axis. The central half-angle is optimized to permit the use of rate gyros with identical sensing range capability despite the fact that the maximum rates about each of the aircraft's orthogonal body axes differ. Other criteria for cone orientation and cone central half-angle selection are to:

- Minimize Analog-to-Digital (A/D) resolution requirements.
- Minimize the average total error amplification due to skewing.
- Retain the quantization necessary to satisfy FCS and Strapdown Attitude and Heading Reference System (SAHRS) requirements.

With careful selection of the cone central half-angle, based on the knowledge of the maximum rate requirements of the aircraft, it is possible to maintain the accuracy of the cone configuration equivalent to that of the conventional, orthogonally mounted, 3-axis rate gyro package. In Figure 10 the apex of the cone is pointed forward. It may also be pointed rearward as shown in Figure 1.

The rate sensed by each gyro is a function of the following:

- Gyro orientation angle
- Gyro central half-angle
- Rates about the aircraft body axes
- Rates induced by the fuselage bending modes and the vibrations and bending modes of the mounting bulkhead

The following equation, in matrix form, describes the actual signal from each rate gyro:

$$\begin{bmatrix} \omega_1 \\ \omega_2 \\ \omega_3 \\ \omega_4 \\ \omega_5 \\ \omega_6 \end{bmatrix} = \begin{bmatrix} \sin \beta_q & 0 & \cos \beta_q \\ -\sin \beta_q & 0 & \cos \beta_q \\ \sin \beta_q \cos \alpha_q & -\sin \beta_q \sin \alpha_q & \cos \beta_q \\ -\sin \beta_q \cos \alpha_q & \sin \beta_q \sin \alpha_q & \cos \beta_q \\ \sin \beta_q \cos \alpha_q & \sin \beta_q \sin \alpha_q & \cos \beta_q \\ \sin \beta_q \cos \alpha_q & -\sin \beta_q \sin \alpha_q & \cos \beta_q \end{bmatrix} \begin{bmatrix} \omega_{zT} \\ \omega_{yT} \\ -\omega_{xT} \end{bmatrix} \quad \text{Equation 1}$$

where:

α_q = to the gyro orientation angle

β_q = to the gyro central half-angle

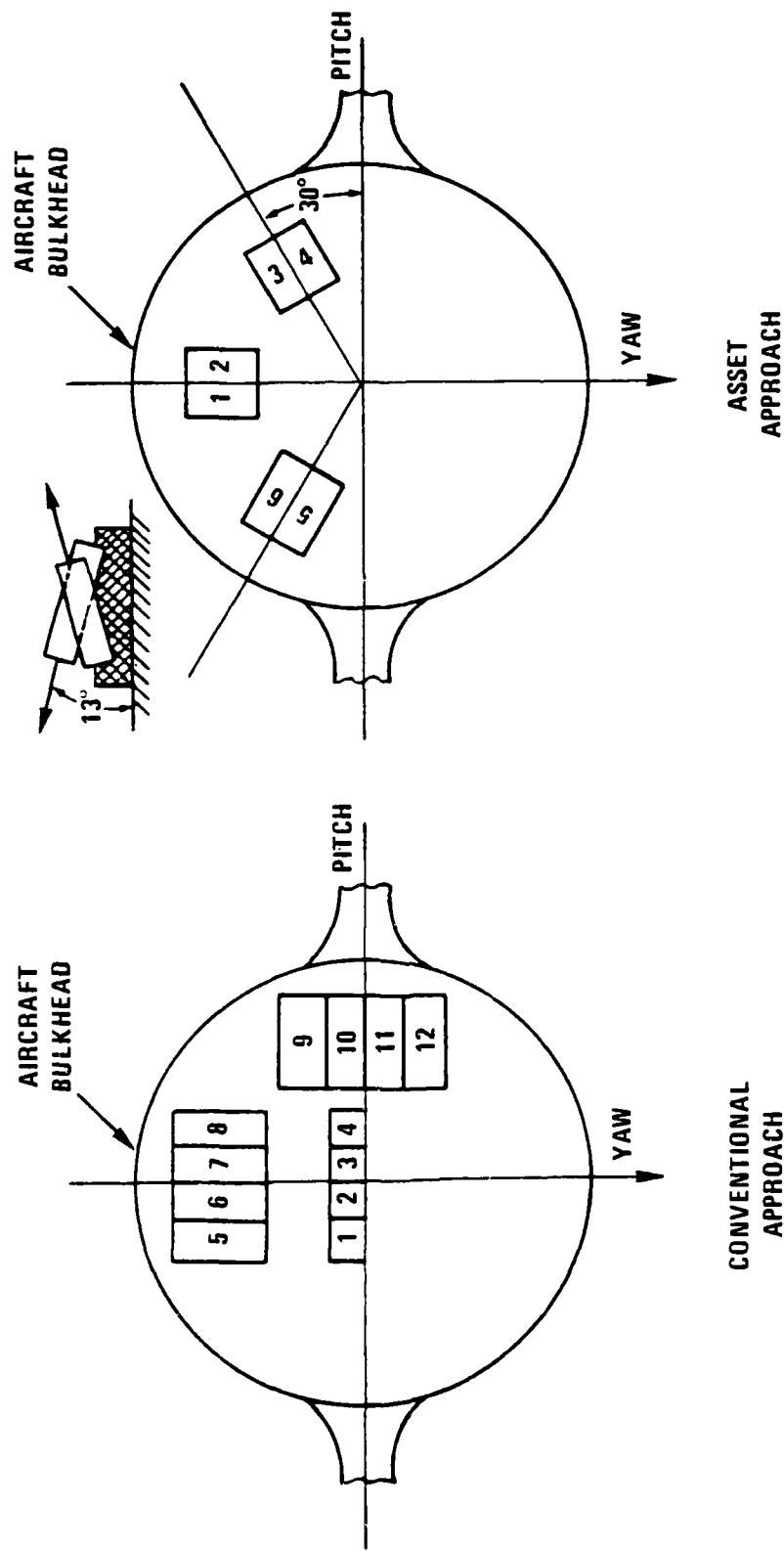


Figure 9. Rate Gyro Configurations for Dual Fail-Operational Capability

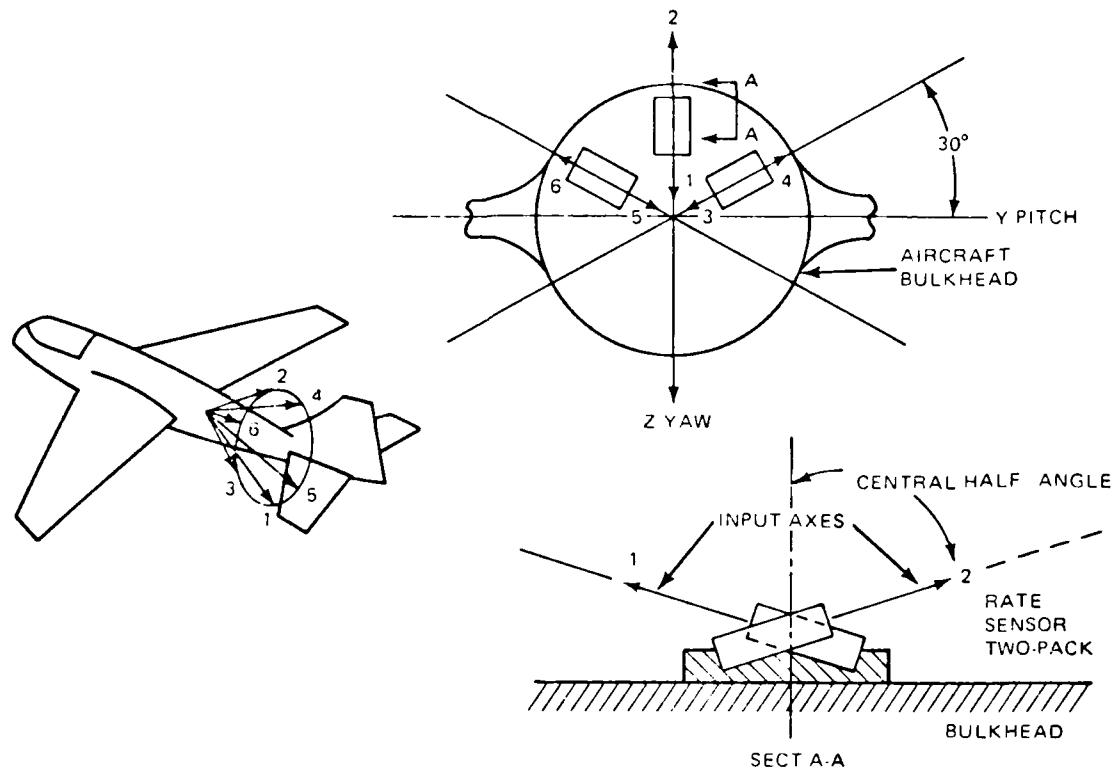


Figure 10. Geometry of Rate Gyro Sensing Axes for the ASSET Concept

ω_i = to the rate sensed by the i th gyro ($i = 1$ to 6)

ω_{zT} = to the total rate about Z axis

ω_{yT} = to the total rate about Y axis

ω_{xT} = to the total rate about X axis

(See Figure 11 for the geometry of the skew angles)

ω_{zT} , ω_{yT} , and ω_{xT} , are a function of aircraft body axes rates, fuselage bending, and bulkhead bending and vibrations. The errors introduced by the structural bendings and vibrations are removed by the ISS Inertial Sensor Data Handling System and the true values of the aircraft rigid body axes rates ω_z , ω_y , and ω_x are made available to the FCS and for attitude and heading computations. These tasks are performed in the digital computer complex with the true values of the aircraft's body rates being updated at the rate of about 50 iterations per second.

The ISS inertial sensor set utilizes six skewed linear accelerometers of which the input or sensing axes form a cone surface oriented along the aircraft's Z body axis as shown in Figure 1. The criteria for cone orientation and cone half-angle selection are the same as those for the skewed rate gyros.

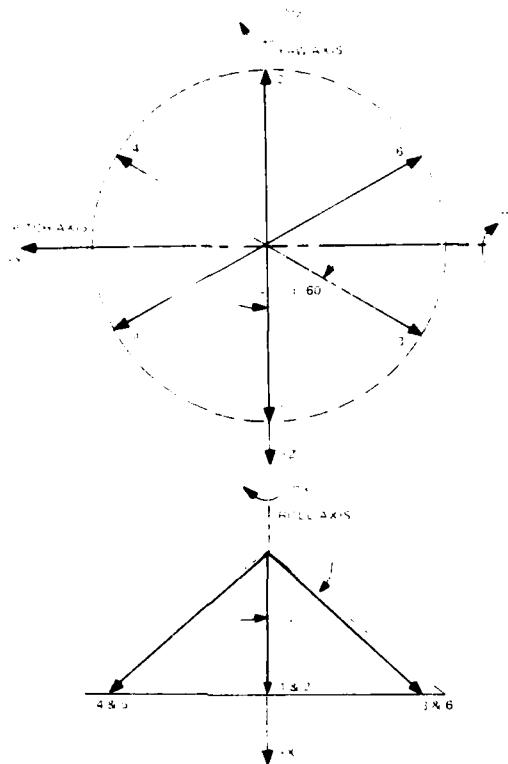


Figure 11. Geometry of a Conical Array of Six Rate Gyros

The linear acceleration sensed by each accelerometer is a function of the following:

- Accelerometer orientation angle.
- Accelerometer central half angle.
- Accelerations along the body axes.
- Length of moment arm from aircraft C.G. to the accelerometer.
- Length of moment arm from center of rotation of the bulkhead to the accelerometer.
- Aircraft angular rates and angular accelerations about the body axes.
- Flexing and bending mode characteristics of the mounting bulkhead and fuselage.

The following equation, in matrix form, describes the actual signal from each linear accelerometer:

$$\begin{bmatrix} a_1 \\ a_2 \\ a_3 \\ a_4 \\ a_5 \\ a_6 \end{bmatrix} = \begin{bmatrix} -\cos \beta_a & 0 & \sin \beta_a \\ -\cos \beta_a & 0 & -\sin \beta_a \\ -\cos \beta_a & -\cos \phi_a \sin \beta_a & \sin \phi_a \sin \beta_a \\ \cos \beta_a & -\cos \phi_a \sin \beta_a & \sin \phi_a \sin \beta_a \\ -\cos \beta_a & \cos \phi_a \sin \beta_a & \sin \phi_a \sin \beta_a \\ \cos \beta_a & \cos \phi_a \sin \beta_a & \sin \phi_a \sin \beta_a \end{bmatrix} \begin{bmatrix} a_{zT} \\ a_{yT} \\ -a_{xT} \end{bmatrix} \quad \text{Equation 2}$$

where:

- ϕ_a = to the accelerometer orientation angle
- β_a = to the accelerometer central half-angle
- a_i = to the linear acceleration sensed by the i th accelerometer ($i = 1$ to 6)
- a_{zT} = to the total acceleration along the Z axis
- a_{yT} = to the total acceleration along the Y axis
- a_{xT} = to the total acceleration along the X axis

(See Figure 12 for the geometry of the skew angles)

a_{zT} , a_{yT} , and a_{xT} are functions of accelerations along the aircraft body axes, aircraft angular rates and angular accelerations about the body axes, and flexing and bending mode characteristics of the mounting bulkhead and fuselage. The errors introduced in these terms are removed by the DHS and the true values of a_x , a_y , and a_z are made available to the FCS, SAHRS, and other using sub-systems at the rate of about 50 iterations per second.

The rate gyros and the linear accelerometers of an ISS are dispersed in order to enhance its survivability in combat operations. Rate gyro and linear acceleration information used for self-contained failure detection and isolation, flight control feedback and for attitude reference purposes must be adjusted to account for rigid body aircraft motion (about the center of gravity) and flexible fuselage

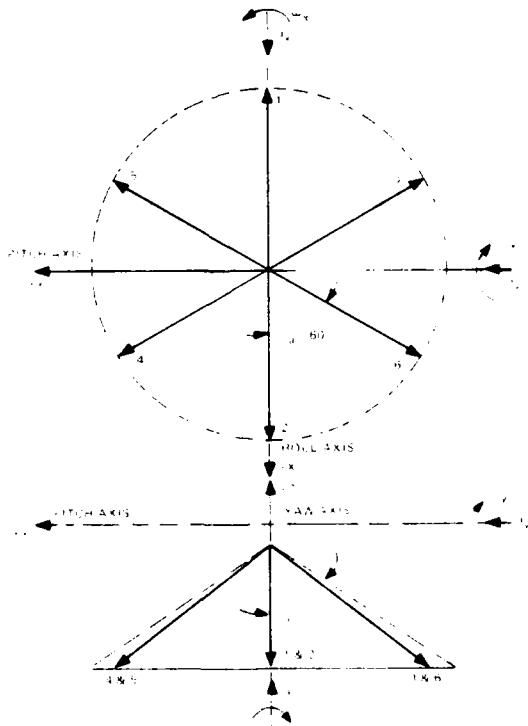


Figure 12. Geometry of a Conical Array of Six Linear Accelerometers

and bulkhead motions. Centripetal and tangential acceleration "errors" that arise from rigid and flexible body motion must be "normalized" or removed from the outputs of hard mounted linear accelerometers (located off of the C.G.) before the information is used for failure detection and isolation, flight control functions or as inputs to the SAHRS.

With flexible inputs not removed from the rate gyro outputs prior to entering the SAHRS routines, the indicated attitude will reflect flexible body as well as rigid body motion.

To minimize errors in estimating the centripetal ($\omega^2 r$) and tangential ($r \dot{\omega}$) accelerations for SAHRS data, the concept of a virtual NAV-BASE and digital bending filter incorporated within the DHS, was developed. The virtual NAV-BASE eliminates the requirements for accurately locating the linear accelerometers with respect to the C.G. of the aircraft. The digital bending filter allows differing bending inputs, sensed by the dispersed linear accelerometers, to be precisely removed with minimum effect in system bandwidth. For flight control usage, these acceleration data, referenced to a virtual NAV-BASE, are then "translated" to the C.G., or to any other location in the aircraft that satisfies control law requirements (i.e., under the cockpit for ride control).

Table 2 (obtained from Reference 1) lists the adjustments required to account for rate gyro and linear accelerometer dispersion.

Figure 13 shows a block diagram of the ISS Inertial Sensor Data Handling System (DHS) which is imbedded as software in the digital computer complex.

Inputs to the DHS are provided by the six rate gyros and the six linear accelerometers. The signal flows from the rate gyros, their normalization, redundancy management, and subsequent distribution will be described first.

Table 2. Adjustments Required to Account for Rate Gyro and Linear Accelerometer Dispersion

	NORMALIZATION FOR FAILURE DETECTION AND ISOLATION (1)	REMOVAL OF FLEXIBLE AND RIGID BODY INPUTS FOR FLIGHT CONTROL (2)	REMOVAL OF FLEXIBLE AND RIGID BODY INPUTS FOR SAHRS (3)
GYRO	NOT REQUIRED IF GYROS ARE ON IN FLEXIBLE BULKHEAD*	NOTCH FILTERS AFTER VOTING	NOT REQUIRED
ACCELEROMETERS	RIGID BODY: ANGULAR RATE (ω) AND ESTIMATE OF ANGULAR ACCELERATION ($\dot{\omega}$) NECESSARY FOR $\omega^2 r$ AND $r \dot{\omega}$ TERMS** FLEXIBLE BODY: NOTCH FILTERS FOR EACH SENSOR PRIOR TO VOTING	→	→

*INFLEXIBLE AT BODY BENDING FREQUENCIES, HIGHER FREQUENCIES HANDLED BY ANALOG PRE-AMPLIFIER IN I/O

** r IS DISTANCE FROM ACCELEROMETERS TO VIRTUAL NAV BASE FOR COLUMNS 1 AND 3 AND DISTANCE TO CENTER-OF GRAVITY FOR COLUMN 2

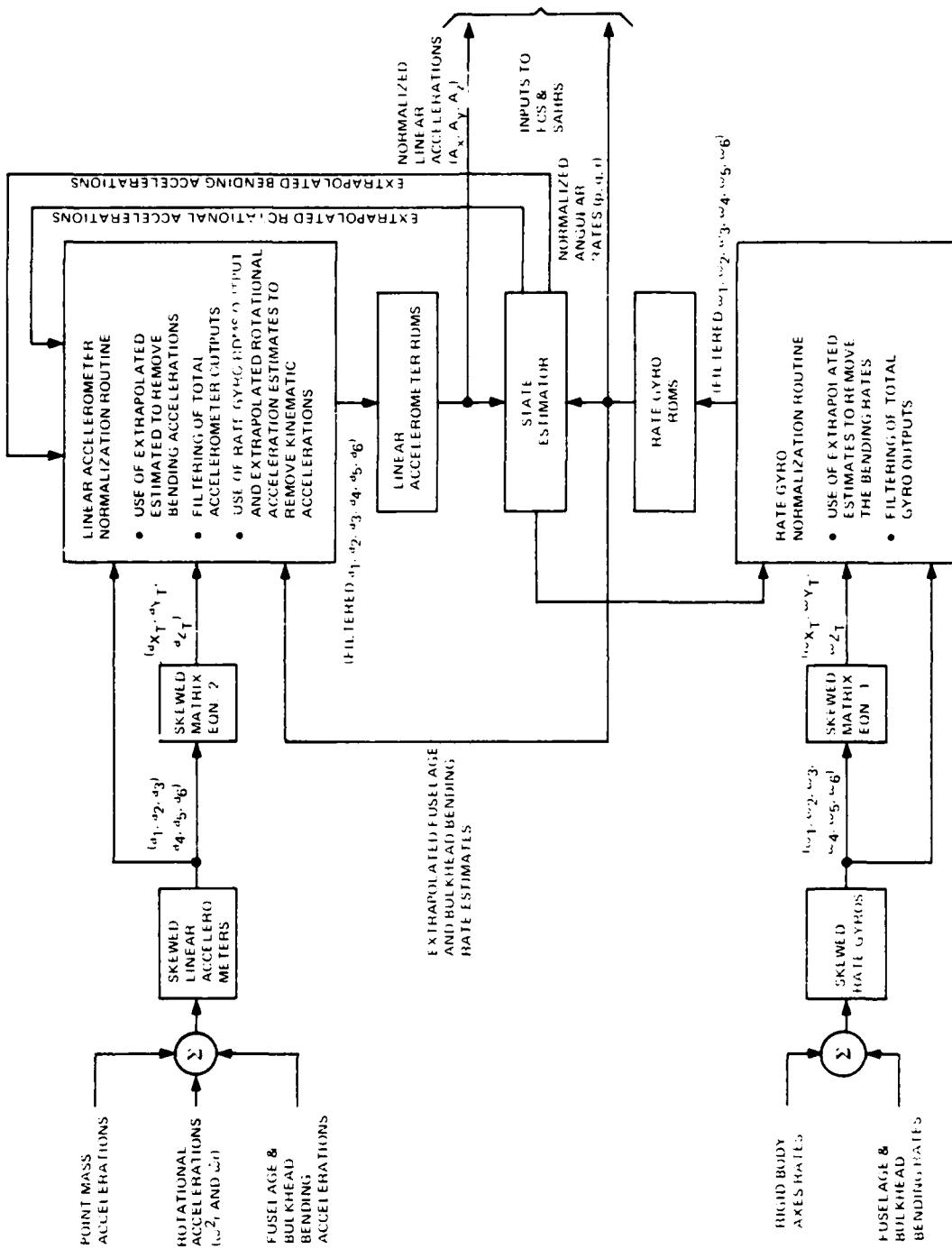


Figure 13. ISS Inertial Sensor Data Handling System (DHS)

The inputs to the skewed rate gyros are the rigid body axes rates (p , q , r) and the undesired fuselage and bulkhead bending rates. Equation 1 is the rate gyro skew matrix. The six simultaneous linear equations are solved at the iteration rate of approximately 50 times per second to obtain updated values of ω_{xT} , ω_{yT} , and ω_{zT} . Extrapolated fuselage and bulkhead bending rate estimates from the State Estimator are subtracted or filtered from ω_{xT} , ω_{yT} , and ω_{zT} and the resultant values are used to aid in the rate gyro normalization process. The angular rates sensed by each rate gyros input axis (ω_1 , ω_2 , ω_3 , ω_4 , ω_5 , ω_6) are also sent to the Rate Gyro Normalization Routine for filtering. The filtered values are used as an aid in the failure detection processes, and calculation of p , q , and r , and are inputted to the Rate Gyro RDMS (See Figure 14). Imbedded within the Rate Gyro RDMS are the following routines:

- Transient Failure Removal Routine (TFRR)
- Sensor Voting Selection Routine (SVSR)
- Failure Isolation Computation Routine (FICR)

The Transient Failure Removal Routine (TFRR) accepts four types of inputs. These are as follows:

- (1) Rate gyro output signals.
- (2) Estimated rate gyro outputs.
- (3) Failure Isolation Computation Routine (FICR) failure signal.
- (4) In-flight BIT (IFBIT) failure signal.

Each rate gyro (after filtering) furnishes an output signal corresponding to the angular rate applied about its input axis. The maximum rates expected from each rate gyro depends on the cone half-angle and on the maximum roll, pitch, and yaw angular rate capabilities of the aircraft. The estimated outputs, one for each rate gyro, are available from FICR computations carried out during the previous iteration. Whenever a rate gyro output is unacceptable for a sufficient period to establish it as permanently failed the FICR makes the information available to the TFRR so that the failed rate gyro is not included in any future calculations.

During every iteration each rate gyro output signal is subtracted from the estimated output. The difference is compared with the threshold corresponding to each rate gyro. If this difference is less than the threshold the output signal is changing at an acceptable rate and the rate gyro is considered to have no temporary failure. If the difference is greater than the threshold the rate gyro is classified as temporarily failed and a signal is made available to notify the Sensor Voting Selection Routine (SVSR) and the FICR. The signal also identifies which rate sensor has failed.

The SVCR produces satisfactory estimates of the angular rates about the roll, pitch, and yaw axes of the aircraft. The voting subroutines (i.e., VOTE 6, VOTE 5, VOTE 4, and VOTE 3) which compute the angular rates are selected on the basis of the failure state.

When there are no failures, the six rate gyro outputs exceed the number required to obtain satisfactory values of p , q , and r . To minimize computational resource requirements VOTE 6 has been designed to select 4 rate gyros, using parity equations, so that the error in the estimate of aircraft rates is minimized. To eliminate the first rate gyro these equations compare each rate gyro output with the estimated value of its output, obtained from the other three rate gyros. The rate gyro output which deviates most from its estimated value is not used in the calculation of p , q , and r . The additional rate gyro is removed by selecting a favorable configuration that minimizes the error in the rate output.

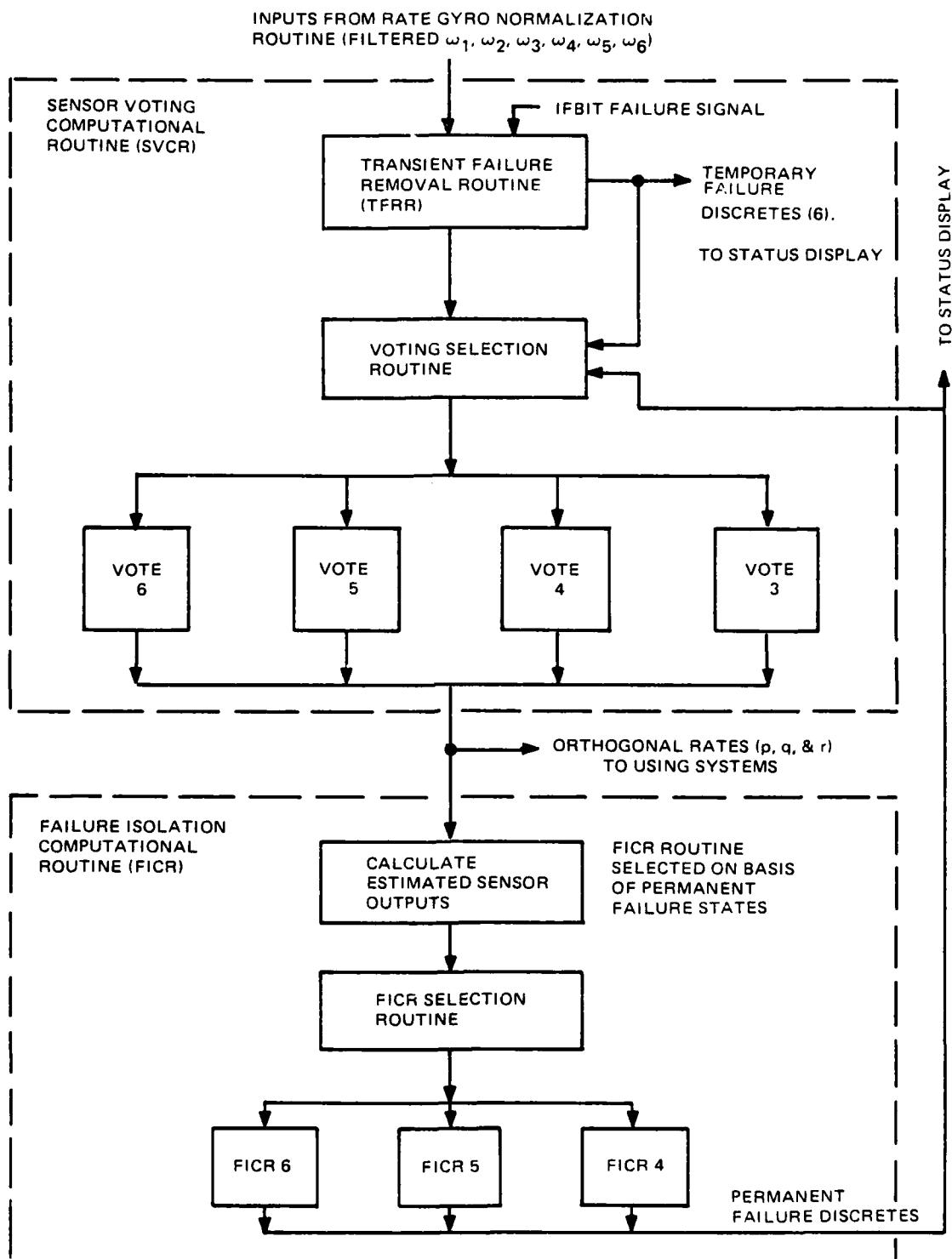


Figure 14. Rate Gyro Redundancy Data Management System (RDMS)

When one sensor has been classified as a failure, VOTE 5 selects the best four of the remaining five by the use of parity equations. With two failures no choice of rate gyros that would decrease roll, pitch, and yaw error is possible and, therefore, the four remaining are used to compute the angular rates in VOTE 4.

The occurrence of three failures leaves only three rate gyro outputs available to calculate p, q, and r in VOTE 3. Under this condition a warning signal is generated to inform the pilot of the potential danger.

Descriptions of the computational processes of the four VOTE subroutines are contained in Appendix 2 of Reference 4.

The Failure Isolation Computational Routine (FICR) detects and identifies rate gyros which have permanently failed. Two ways of examining for permanent failures are used. First, if any of the rate gyro estimated outputs exceed 5 degrees per second, the error of those rate gyros are examined using a Cross Voter Comparison Monitoring (CVCM) technique while the other rate gyros are not tested. If all of the rate gyro estimated outputs are less than 5 degrees per second the errors from all of the rate gyros are estimated using a Cross Channel Comparison Monitoring (CCCM) technique.

Using the CVCM technique, the values of the p, q, and r rates produced by the SVCR are used to generate estimates of the six rate gyro outputs. The individual estimates of the outputs are then divided by the actual rate gyro outputs. If the absolute value of (dividend - 1.0) is the largest of all being examined and greater than a threshold for a period of 0.2 seconds, the rate gyro is classified as a permanent failure.

With the CCCM technique, used when the rate gyro outputs are less than 5 degrees per second, the error of a rate gyro is estimated by means of one of a set of fifteen equations (called "parity equations") which are defined and described in Reference 4. Briefly the fifteen parity equations describe the relationships between all possible groups of four rate gyros selected from the six in the skew arrangement.

When the parity equations are all equal to zero the rate gyro outputs are error free. However, an equation not equal to zero indicates an error and a possible permanent failure. If the error persists continuously for at least 1 second, the rate gyro producing the error is classified as permanently failed.

With no permanent failures subroutine FICR 6 is used. With one permanent failure subroutine 5 is used. When none or one previous permanent failure exists, the next permanent failure can be identified by the parity equations. With two permanent failures (FICR 4), the parity equations indicate that one of the four remaining rate gyros is a permanent failure but it cannot identify the one that has failed.

The FICR subroutines are defined and described in detail in Reference 4.

Existing outside the functional boundaries of the RDMS, the IFBIT detects catastrophic (permanent) failures of the rate gyros. The IFBIT output furnishes information that a rate gyro has failed and its identity.

The inputs to the skewed linear accelerometers are the point mass accelerations, and the undesired rotational, fuselage and bulkhead bending accelerations. Equation 2 is the linear accelerometer skew matrix. The six simultaneous linear equations are solved at the iteration rate of approximately 50

times per second to obtain updated values of a_{xT} , a_{yT} , and a_{zT} . Extrapolated fuselage and bulkhead bending accelerations and rotational accelerations from the State Estimator, and the p, q, and r angular rates from the Rate Gyro RDMS are used to remove the errors in a_{xT} , a_{yT} , and a_{zT} and the resultant values are used to aid in the linear accelerometer normalization process. The linear accelerations (a_1 , a_2 , a_3 , a_4 , a_5 , a_6), sensed by each linear accelerometer, are sent to the Linear Accelerometer Normalization Routine for filtering. The filtered values are used as an aid in the failure detection and isolation process, and the calculation of a_x , a_y , a_z , and are inputted to the Linear Accelerometer RDMS (see Figure 15). The calculation processes, logic, and signal flow are the same as those for the Rate Gyro RDMS.

The State Estimator of the ISS Inertial Sensor DHS is used to calculate an estimate of the aircraft's fuselage and bulkhead bending rates and accelerations as well as the kinematic accelerations at the dispersed sensor locations. The estimates of bending and kinematic effects are then subtracted from the raw sensor data to permit effective failure detection and isolation and accurate calculation of the aircraft's orthogonal angular rates and linear accelerations.

State-of-the-art digital design techniques were applied by the Grumman Aerospace Corporation to develop the State Estimator. Reference 6 describes the development and functions of the State Estimator.

3. ISS/SAHRS Function for Sensing Attitudes, Heading, and Inertial Velocity Components

A Strapdown and Heading Reference System (SAHRS) is integrated within the ISS for the calculation of the following parameters using corrected values of p, q, and r and a_x , a_y , and a_z and inputs from non ISS sensors:

- Roll altitude (Φ)
- Pitch altitude (θ)
- Heading altitude (Ψ)
- North component of velocity
- East component of velocity
- Vertical component of velocity

Corrected values of the p, q, and r angular rates, obtained from the ISS Inertial Sensor DHS are integrated once and processed to obtain the Euler Angles Φ , θ , and Ψ . The corrected values of a_x , a_y , and a_z orthogonal linear accelerations, also obtained from the DHS, are integrated once to obtain the inertial velocity components along the aircraft body axes and these values together with the Euler Angles are used to obtain the North, East, and Vertical inertial velocity components. The integrated values of these velocity components are used to calculate position changes for navigational purposes.

The outputs of the ISS/SAHRS are used by the following aircraft systems:

- Flight Control System (FCS)
- Automatic Flight Control System (AFCS)
- Cockpit displays
- Weapons delivery

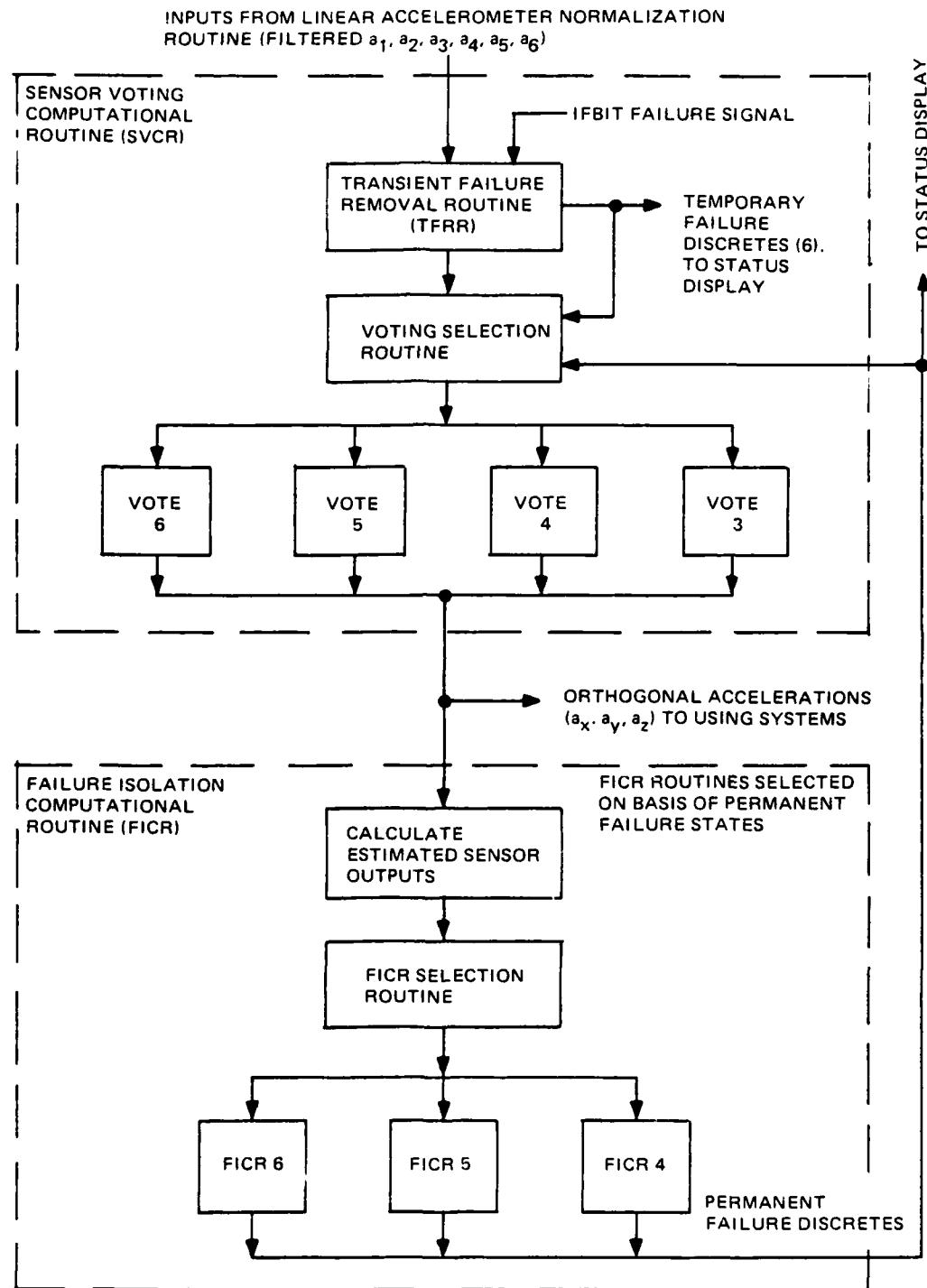


Figure 15. Linear Accelerometer Redundancy Data Management System (RDMS)

Figure 16 shows the ISS/SAHRS Signal Flow Block Diagram. The skewed rate gyro and linear accelerometer data are resolved by the ISS Inertial Sensor DHS, to three outputs along vehicle axes. Spurious inputs from aircraft rigid body and flexible body motions, coupled with the dispersion of skewed vector sensors, are removed by the DHS.

After initialization of the inertial frame to vertical and north the Attitude Transformation Update and Orthonormalization Routine operates on the rate gyro outputs to maintain the knowledge of the angular relationship between the aircraft body and inertial frame. This angular relationship is used to transform the linear accelerometer data obtained along the body axis to the vertical. The information is then mixed with "aiding" data such as air data velocity, to "erect" the system through a long-time-constant filter to the vertical. Therefore, the long-term accuracy of the vertical indication depends on the accuracy of the reference velocity. The reference frame is "slaved" to north, defined by the magnetic heading information, again through a long-time-constant filter.

Relative velocity and relative position (range and range rate) can be obtained from this system when the system is aided with relative position data from a landing guidance sensor such as those described in Reference 7.

Alignment on the ground and in flight will be performed in two steps:

- Coarse Alignment – where one or two iterations of the magnetic heading and accelerometer data are used to generate a crude estimate of attitude and heading (body axes orientation with respect to vertical and north). After this is completed, the system reverts to Fine Alignment.
- Fine Alignment – is composed of a 40-sec time constant, second-order vertical erection loop that uses reference velocity, and a 10-sec time constant, second-order heading filter that utilizes magnetic heading information.

Attitude transformation presents one of the most demanding requirements on the throughput of a digital computation system. Studies utilizing the information in Reference 8 have shown that a 50 msec update rate using a second-order quaternion algorithm will be satisfactory for SAHRS performance. Typically, at a 300 deg/sec roll rate input the computational error is approximately 0.04 degrees per iteration. Subsection 3.4 of Reference 2 shows that this SAHRS 50 msec update rate will satisfy ISS computational requirements.

The results of the SAHRS performance analysis and an equipment survey showed that in-production Single-Degree-Of-Freedom (SDOF) rate integrating gyros in the 12,000 to 30,000 gm-cm/sec angular momentum class and in-production linear accelerometers with self-contained loop electronics will satisfy the performance requirements of an ISS.

Typical rate gyro performance requirements for an ISS/SAHRS are listed as follows:

• Random drift	0.20 deg./hr. (1σ)
• Start/start bias repeatability	0.20 deg./hr. (1σ)
• G sensitivity drift repeatability	0.20 deg./hr./g (1σ)
• Anisoelasticity	0.50 deg./hr./g ²
• Compensated temperature sensitivity	0.006 deg./hr./°C
• Wheel run-up time	3 seconds
• Scale factor linearity	200 PPM (parts per million)

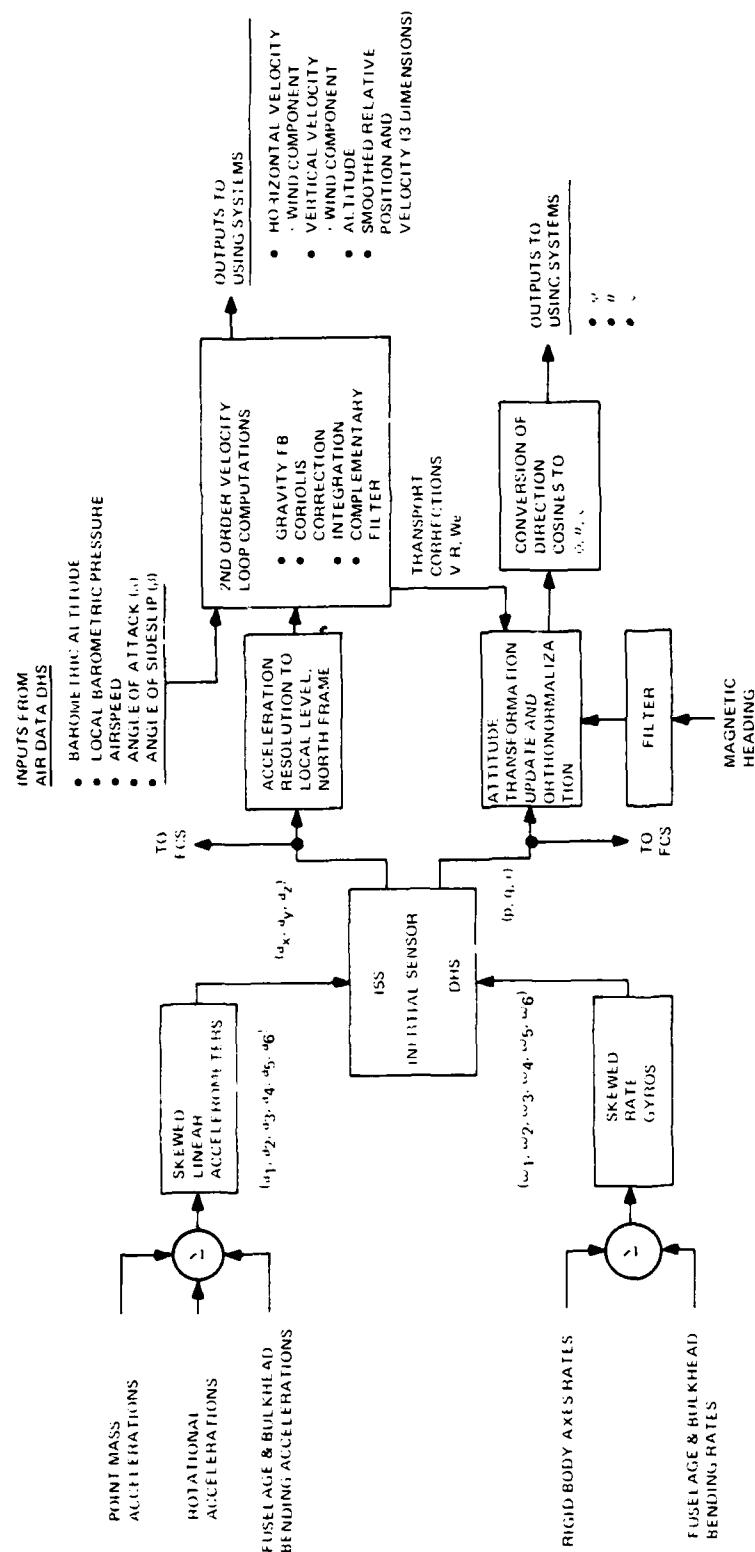


Figure 16. ISS/SAHRS Signal Flow Block Diagram

Overall SAHRS performance and accuracy requirements are listed as follows:

● Magnetic Heading	0.5 degree (1σ)
● Pitch	0.15 degree (1σ)
● Roll	0.15 degree (1σ)
● Body rates	0.5%
● Linear acceleration	0.005 g

(Note: The above performance and accuracy values are attained by the Sperry SRS-1100 SAHRS)

The performance requirements for the ISS/SAHRS inertial sensors are dictated by the accuracies required for inputs to the FCS of a VTOL type aircraft designed to land on small ship platforms during adverse weather and sea state conditions.

ISS REDUNDANT INPUT/OUTPUT (I/O) SYSTEM

The redundant and dispersed Input-Output (I/O) system is the element of the ISS concept that links the data from the redundant sensor set to the using systems via a redundant digital computer complex. To reduce vulnerability to combat damage the I/O system signal paths are redundant and dispersed. Figure 17 is a simplified signal flow and block diagram of an ISS I/O system for a DFCS. It represents only one of the many methods in which an I/O system may be configured. Flight safety critical inputs to the digital Flight Control Computer (FCC) complex are transmitted via redundant and dedicated signal paths. Output signals from the FCC's to flight safety critical signals are transmitted in the same manner. Signals from the FCC's to systems which can be in a failed state without degrading the controllability of the aircraft are transmitted via the Dualized Avionics Multiplex Bus.

Note that the outputs of the Air Data Sensors, which are in the digital format, are inputted into the FCC's via dedicated triplex multiplex bus signal paths. Figure 18 shows this interface in greater detail. Each of the three Universal Transducer Modules shown contain a microprocessor for pre-processing the air sensor data and to provide data output control, a power supply, and the I/O electronics necessary to interface with the redundant multiplex buses. Since the air data values are in digital form in the microprocessors their transmission to the FCC's can be most efficiently accomplished via multiplex buses.

The other flight safety critical inputs to the FCC's shown in Figure 17 are in the analog format and are converted to the digital format in the FCC's.

For all inputs the functions of failure detection, failure isolation and signal selection are performed in the FCC's. Signal outputs from the FCC's to the flight safety critical systems (after undergoing D/A conversion) are in the analog format.

Methods of inputting the critical ISS inertial sensor data to the FCC's will now be described. The most reliable method is that of "cross-strapping" the output of each inertial sensor to all three FCC's as shown in Figure 19. The inertial sensors are housed in three Inertial Component Assemblies (ICA's). Each ICA contains two skewed single degree-of-freedom (SDOF) rate gyros and two skewed linear accelerometers. The ICA's are widely dispersed and mounted on one fuselage bulkhead to reduce the vulnerability of the ISS inertial sensors set to combat damage. Loss of an ICA will not prevent the FCC's to output corrected values of p , q , r , a_x , a_y , and a_z to the using systems. The

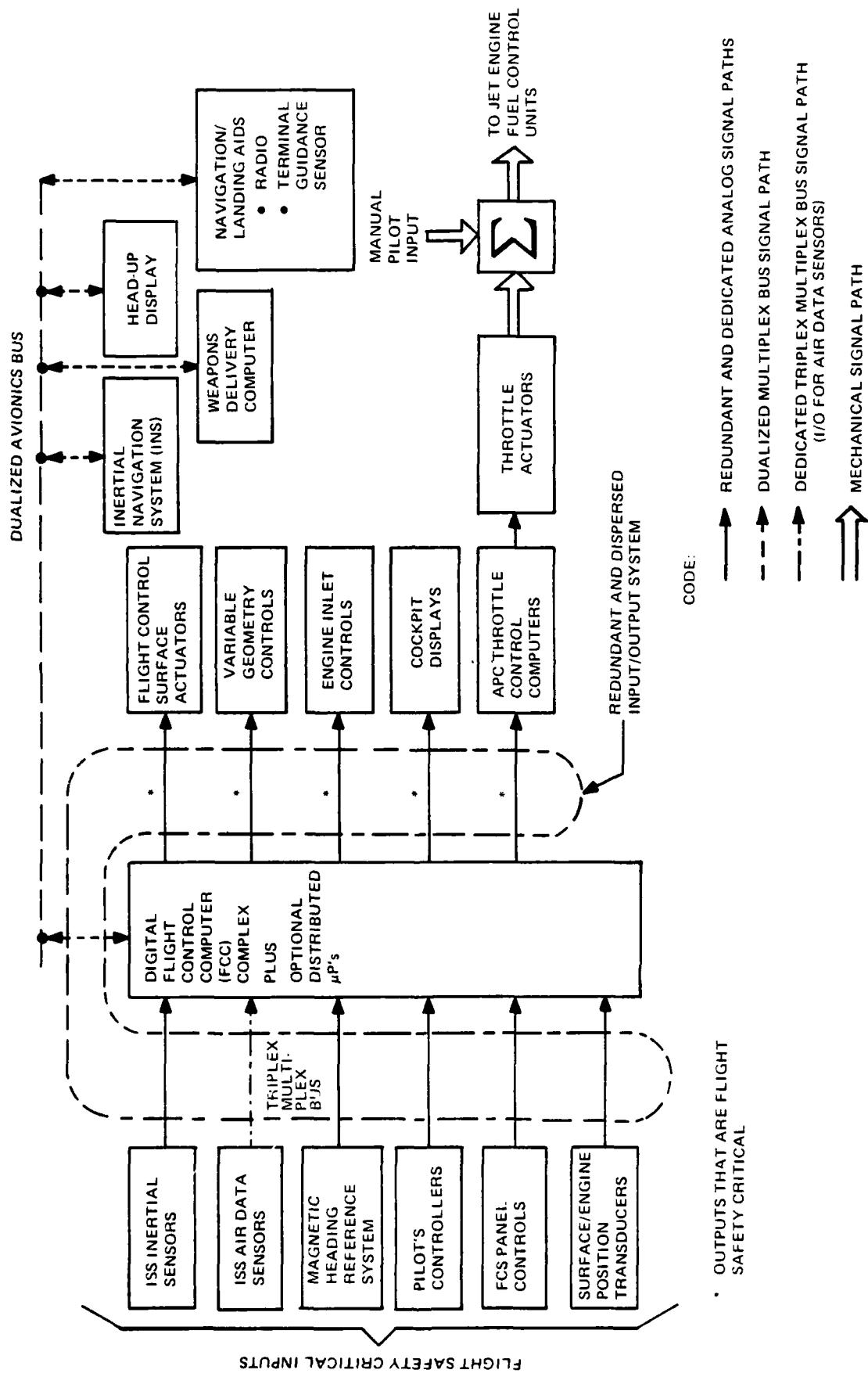


Figure 17. ISS Input/Output System for a Digital Flight Control System (DFCS)

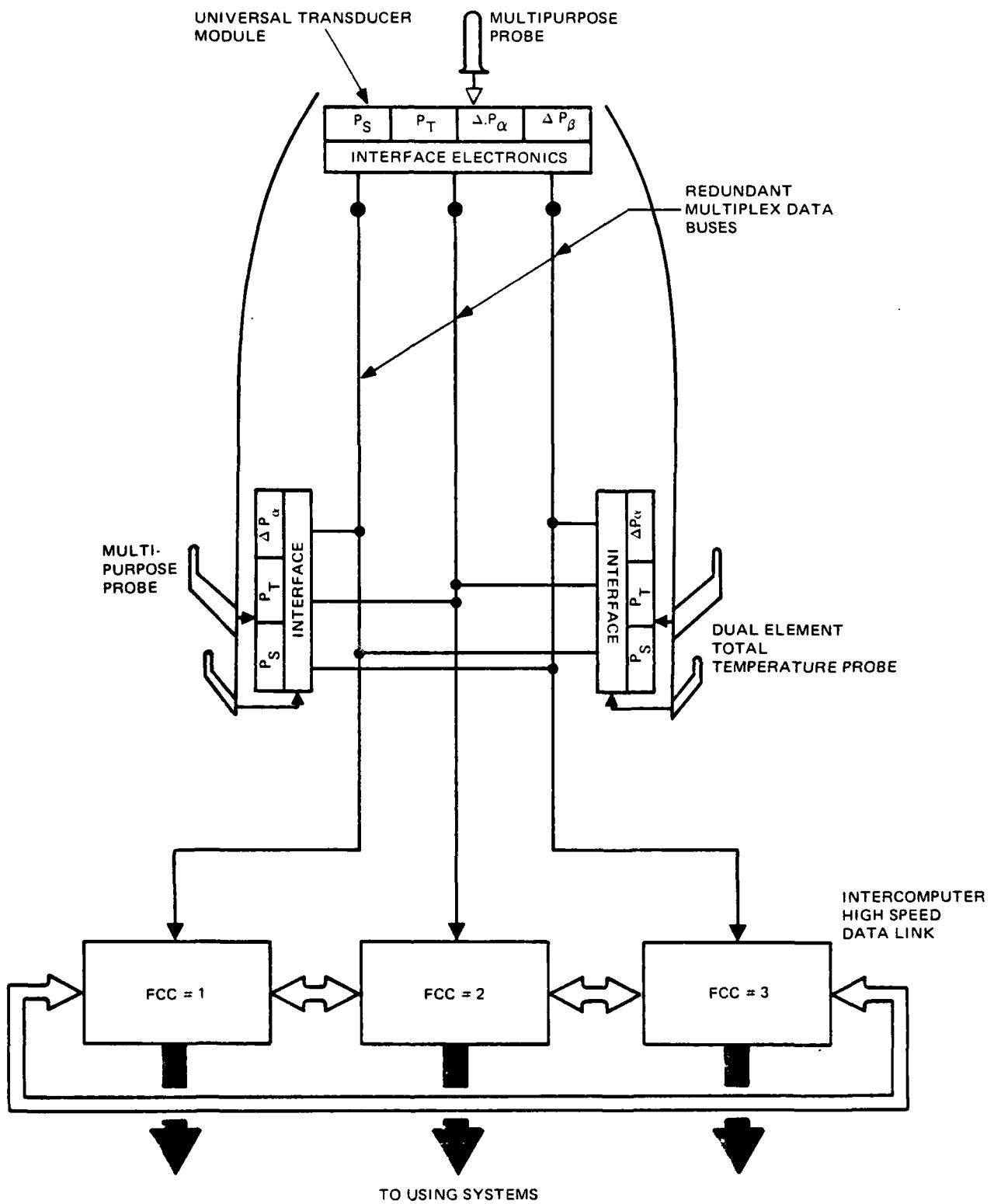


Figure 18. Air Data System/Flight Control Computer Complex Interface

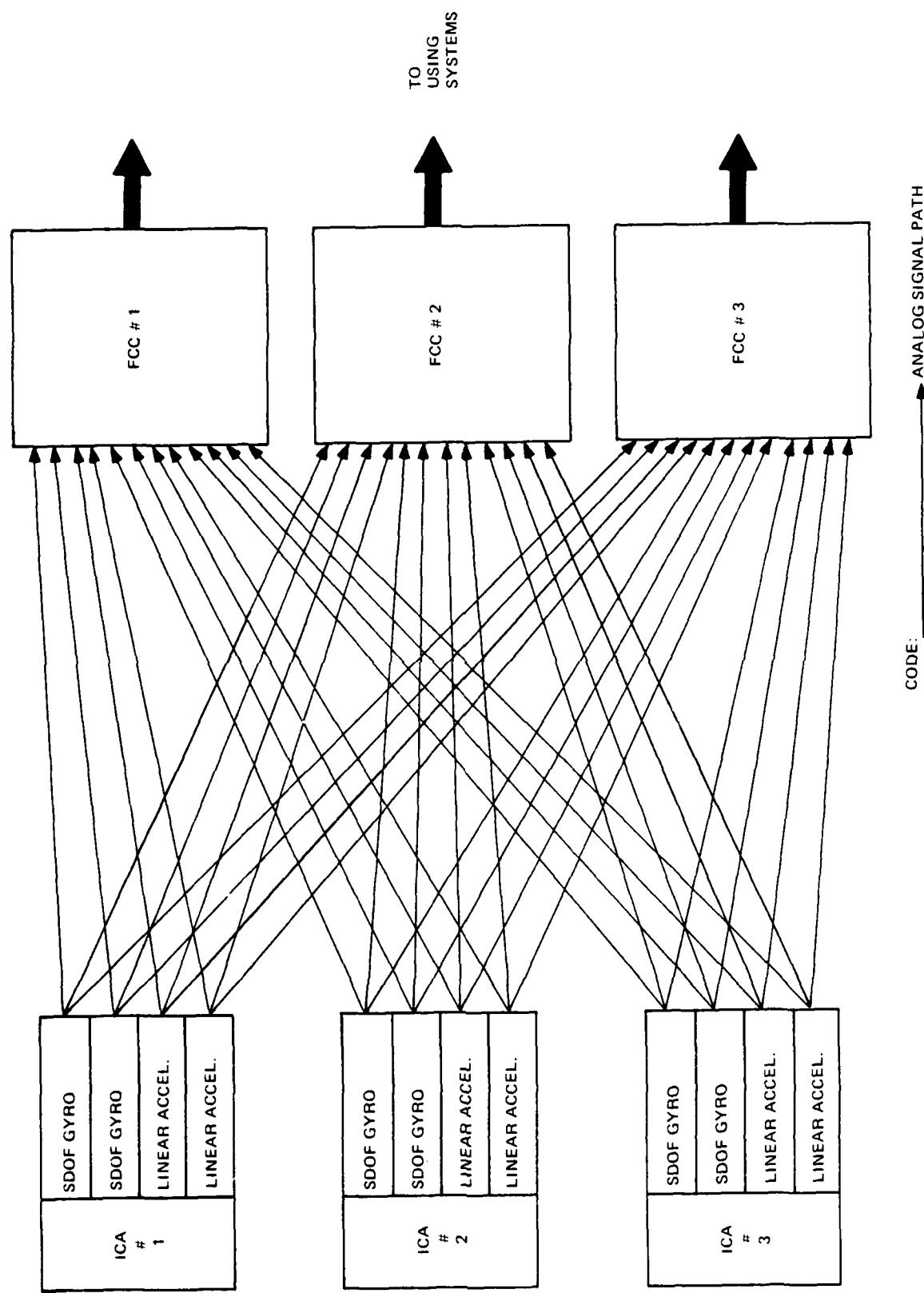


Figure 19. Interface of the Inertial Component Assemblies with the Flight Control Computers Via Cross-Strapping

losses of two like sensors (two rate gyros and/or two linear accelerometers can be readily compensated for by the RDMS of the ISS Inertial Sensor DHS.

The output signals of the inertial sensors are in the analog format and the required A/D conversions are performed in the FCC's. Note that each FCC has twelve dedicated input ports to receive the inertial sensor signals. For each input port there is a dedicated A/D converter. It is readily apparent that the cross-strapping method is costly in terms of system Life Cycle Costs (LCC), hardware parts count, and weight.

Cross-strapping is used in the quadruplex DFCS of the F-18. To attain two fail-operational performance the DFCS utilizes four FCC's, twelve rate gyros, and twelve linear accelerometers (four sensors per axis). The axes of the F-18 inertial sensors are orthogonal. Each FCC has twenty-four dedicated input ports to receive the analog signals of the twelve rate gyros and twelve linear accelerometers. In contrast an ISS concept using cross-strapping requires half as many input ports (12 vs. 24) to obtain two fail-operational performance.

A less costly method of inputting inertia sensor signals is shown in Figure 20. The output of each inertial sensor (in the analog format) is sent to only one of the FCC's and then transmitted to the other two FCC's via the inter computer high speed data link. Since each FCC receives data from all inertial sensors all FCC's are able to perform the ISS Inertial Sensor DHS function. Note that each FCC now requires only four dedicated input ports and A/D converters. This method of inputting inertial sensor data to the FCC's is employed in the triplex DFCS of the AFT1/F-16 aircraft. Studies conducted by the General Dynamics Corporation have shown that this method is less costly than cross-strapping with only a small decrease in reliability.

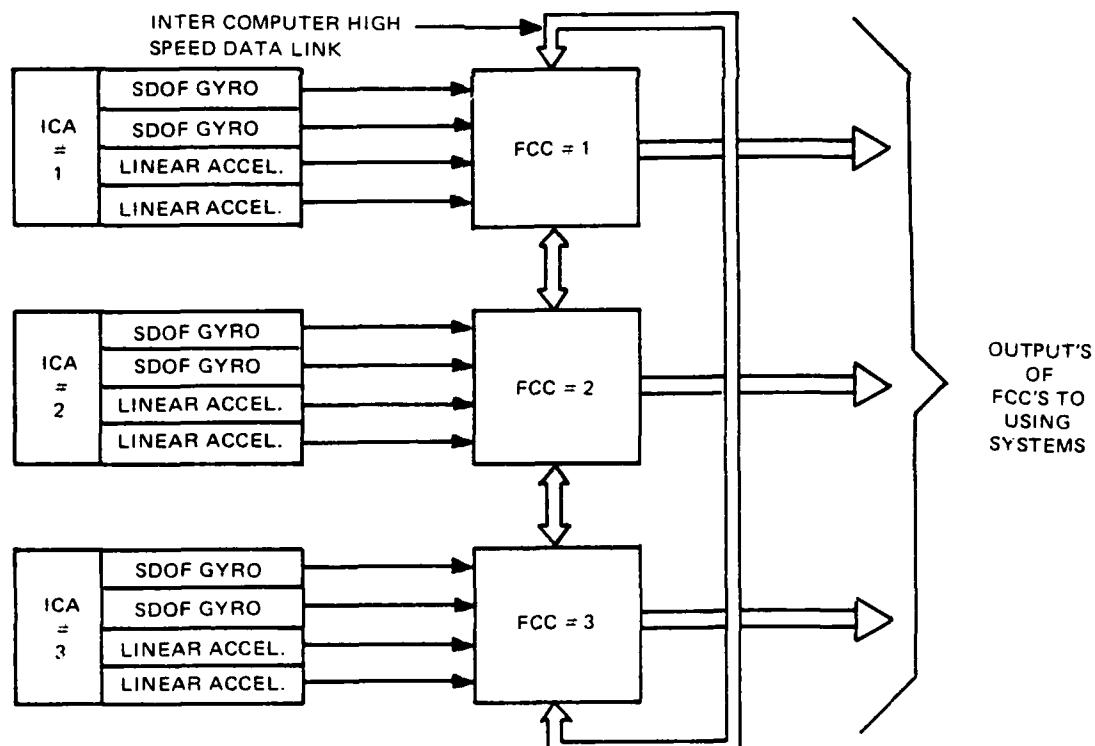


Figure 20. Interface of the Inertial Component Assemblies with the Flight Control Computers Via Inter Computer High Speed Data Link

In some future DFCS configurations it might be desirable not to burden the FCC's with the ISS Inertial Sensor DHS function. This burden could be placed upon a network of distributed microprocessors (μ p's) configured as shown in Figure 21. Each inertial sensor output signal is sent to only one input port of a μ p and then transmitted to the other two μ p's via an inter computer high speed data link. The outputs of the three ICA μ p's are corrected values of p , q , r , a_x , a_y , and a_z which are inputted to the three FCC's.

ISS COMPUTATION NETWORKS WITHIN REDUNDANT DIGITAL COMPUTER COMPLEXES

The use of the ISS concept in aircraft with DFCS's does not impose the requirement for the development and procurement of dedicated or special purpose digital computers. This course of action would be most undesirable as it would greatly increase the Life Cycle Costs (LCC) of operational aircraft integrated with an ISS. The required computations for an ISS may be performed in digital computer architectures listed as follows:

- (a) Use of available on-board redundant FCC's on a time shared basis.
- (b) Use of a computational architecture containing both redundant distributed microprocessors and redundant FCC's on a time shared basis. The microprocessors, dedicated for ISS functions, would reduce the computational load of the FCC's.

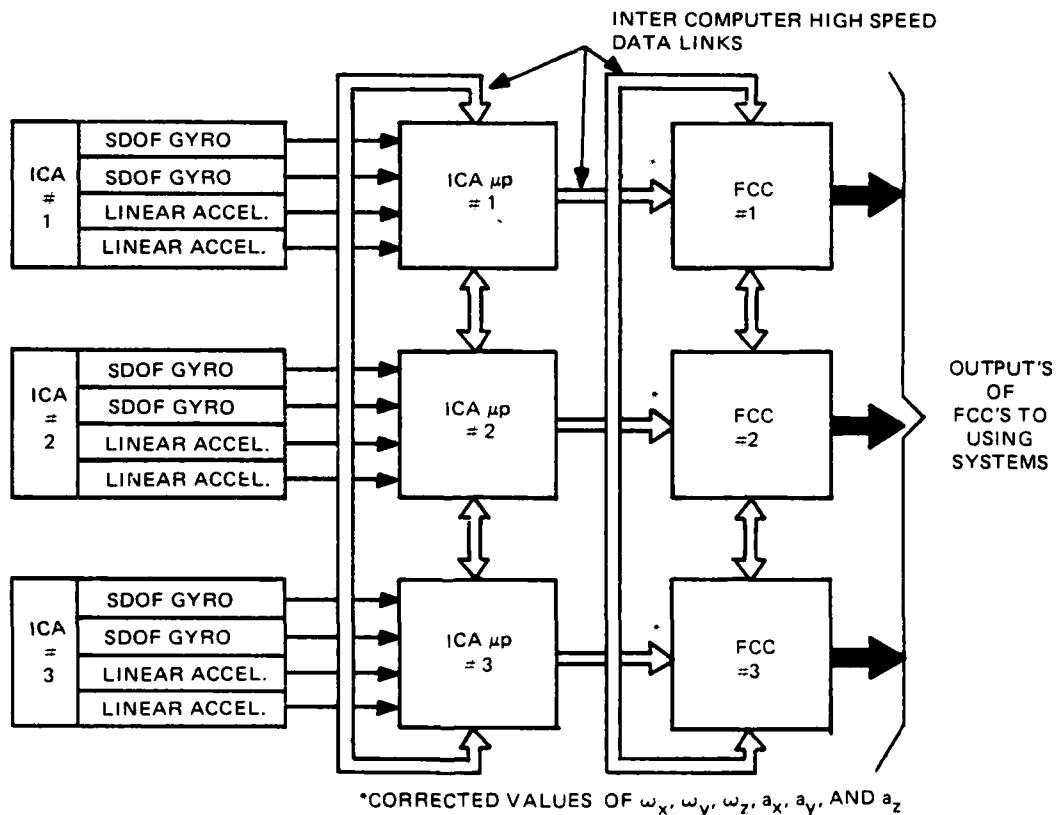


Figure 21. Interface of the Inertial Component Assemblies and ICA μ p's with the Flight Control Computers

THE ADVANCED FIGHTER TECHNOLOGY INTEGRATOR (AFTI/F-16) DEVELOPMENT AND INTEGRATION PROGRAM

The purpose of the Advanced Fighter Technology Integrator (AFTI/F-16) Program is to design, develop, and integrate into a modified F-16 aircraft, a set of advanced technologies capable of providing improved weapon system effectiveness and survivability in air-to-air and air-to-surface. The set of technologies to be designed, developed, integrated, and verified by flight tests in an AFTI/F-16 aircraft are listed as follows:

- Advanced Digital Flight Control System (DFCS)
- Advanced Integrated Fire and Flight Control (IFFC) System
- Advanced Armament/Weapon Delivery System
- Aerodynamic Optimization
- Advanced Crew Stations Systems and Concepts

Figure 22 summarizes the objectives of the AFTI/F-16 Program.

The AFTI/F-16 Program is being sponsored and funded by the Air Force, Navy, and NASA. The prime contractor is the General Dynamics (GD) Corp./ Fort Worth Division. The Navy is providing part of the funding and program direction for only one of the set of technologies being developed; the advanced Digital Flight Control System (DFCS). Only the design and development of this technology will be described in this report. However aircraft subsystems interfacing with the DFCS will also be described. The subcontractor for the design, development, and fabrication of the main components of the DFCS is the Flight Systems Division of the Bendix Corporation located at Teterboro, New Jersey.

The general objectives of the AFTI/F-16 DFCS Advanced Development Program are as follows:

- Demonstrate the operation of a second generation DFCS.
- Evaluate the operational capabilities of a triplex DFCS.
- Demonstrate the benefits of multi-mode flight control capability.
- Determine the configuration of an independent flight control backup system.
- Evaluate pilot acceptance of Direct Lift Control (DLC), Direct Side-Force Control (DSF) and fuselage pointing capabilities.
- Demonstrate increased mission effectiveness and reduced pilot workload.
- Develop a data base and experience for future DFCS applications.

Some of the more specific objectives are as follows:

- Define computer characteristics
 - Architecture
 - Throughput requirements
 - Memory requirements

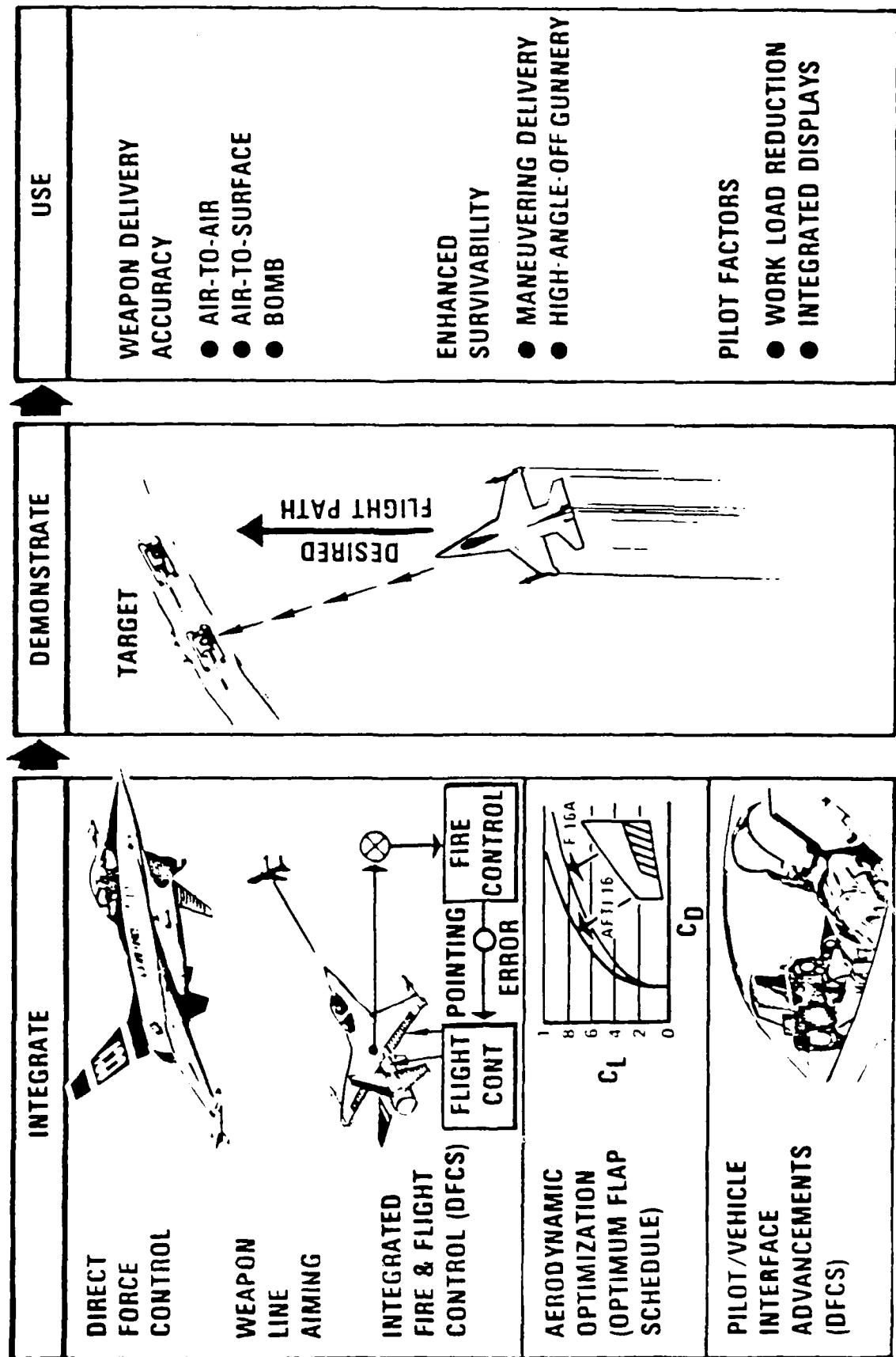


Figure 22. Objectives of the AFTI/F-16 Program

- Evaluate redundancy data management for dual fail-operational (F/O) capability.
- Evaluate confidence level of in-line testing.
- Define independent backup flight control switching intelligence.
- Define 1553 Multiplex Bus interfaces.
- Confirm multi-mode selection
 - Normal
 - Air-to-air
 - Air-to-ground strafing
 - Air-to-ground bombing.

DESCRIPTION OF THE AFTI/F-16 DIGITALFLIGHT CONTROL SYSTEM (DFCS)

The AFTI/F-16 Digital Flight Control System (DFCS) is a full Fly-By-Wire (FBW) system with no provision for mechanical back-up. Advanced digital computational technology, electronic mechanization, and control theory are used to eliminate mechanical type input signal paths in all axes. From Figure 23 it is evident that no mechanical linkages or control cables are used between the cockpit controllers (side-stick, rudder pedals, flight control panel, etc), the control surface and speed brake actuators, and the leading-edge flap power drive unit. The AFTI/F-16 Digital FCS represents a major technological advance over the current analog FBW FCS used in the operational F-16's. A DFCS permits the pilot to choose from a "menu" of multiple, selectable tasks/functions in order to optimize the flight control task as a function of flight condition, aircraft configuration, and mission segment. This fact is illustrated in the block diagram of Figure 24.

Basically the AFTI/F-16 DFCS is configured as a triplex-redundant system with three independent electronic branches. The prime electronic assemblies are the three identical digital Flight Control Computers (FCC's) which perform the functions listed in the schematic of Figure 25. The triplex aircraft motion sensors (rate gyros and accelerometers) provide feedback stabilization and aircraft state information to the FCC's. Quadruplex redundant pilot sidestick and rudder pedal force sensors accept pilot command inputs and transmit them in the form of analog voltage signals to the triplex FCC complex for processing. The reason for using quad-redundant force sensors will be explained later.

The DFCS is protected through automatic failure detection via voting, in-line monitoring, and analytical redundancy techniques, failure isolation, and subsequent system reconfiguration. The triplex DFCS is able to provide two fail-operational performance (within a 95% probability) following two FCC failures.

Figure 25 illustrates schematically the various inputs to the triplex DFCS complex, the values of which are used by the stored control laws to compute control surface deflection commands which are transmitted to the seven Integrated Servoactuators (ISA) via triplex electrical signal paths.

The AFTI/F-16 aircraft is a highly modified version of an F-16A from which the analog FBW type FCS has been removed and replaced with a triplex DFCS. In the design of the F-16 series aircraft advanced flight control technologies were allowed to impact the design process at the beginning. One advanced flight control concept, in particular, that of Reload Static Stability (RSS) had a very large impact on the F-16 design in regard to its resultant aerodynamic configuration, size, weight, range, performance, and thrust to weight ratio. It was the use of RSS that resulted in the selection

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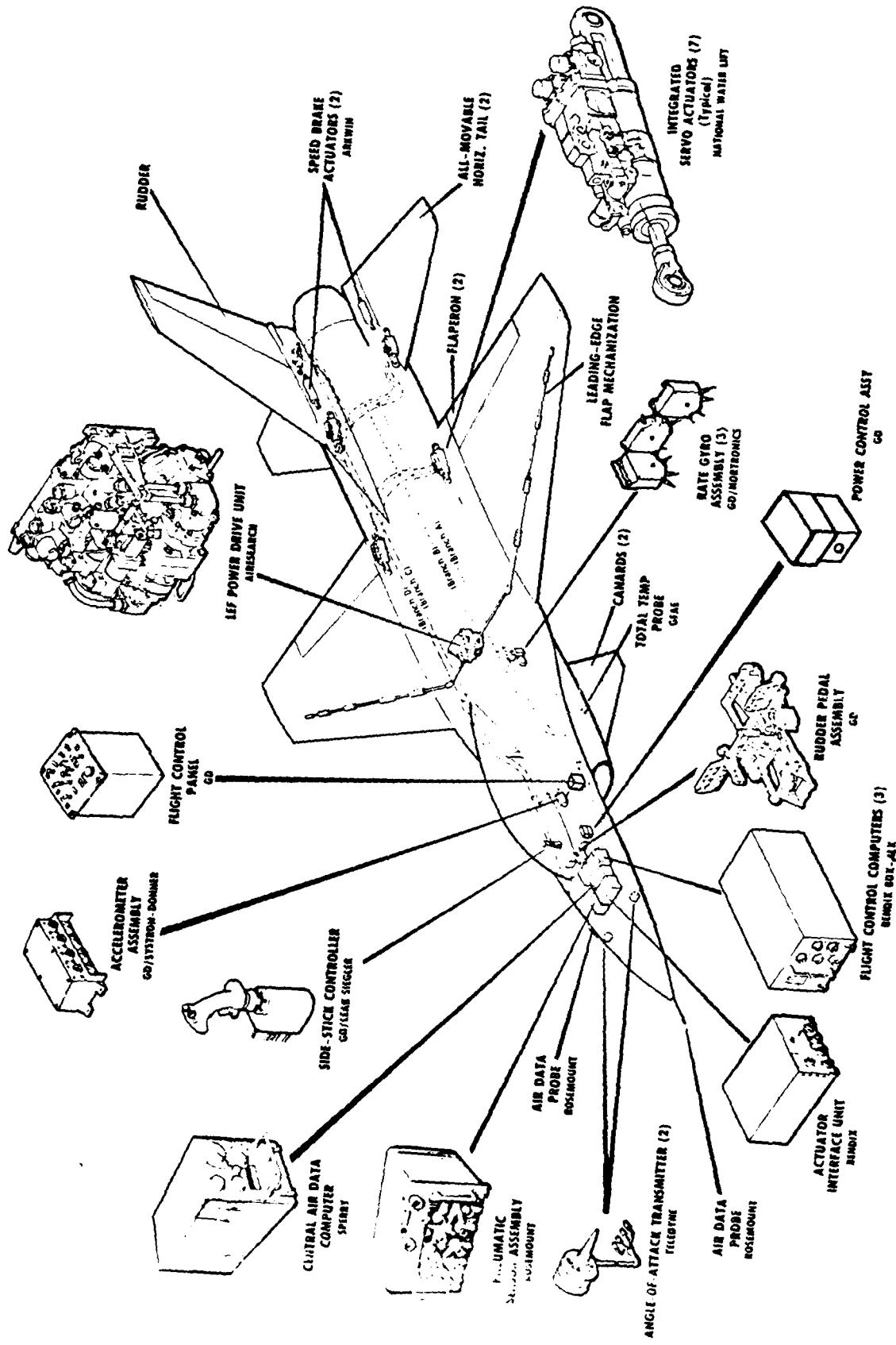


Figure 23. Digital Fly-By-Wire Flight Control System For the AFTI/F-16

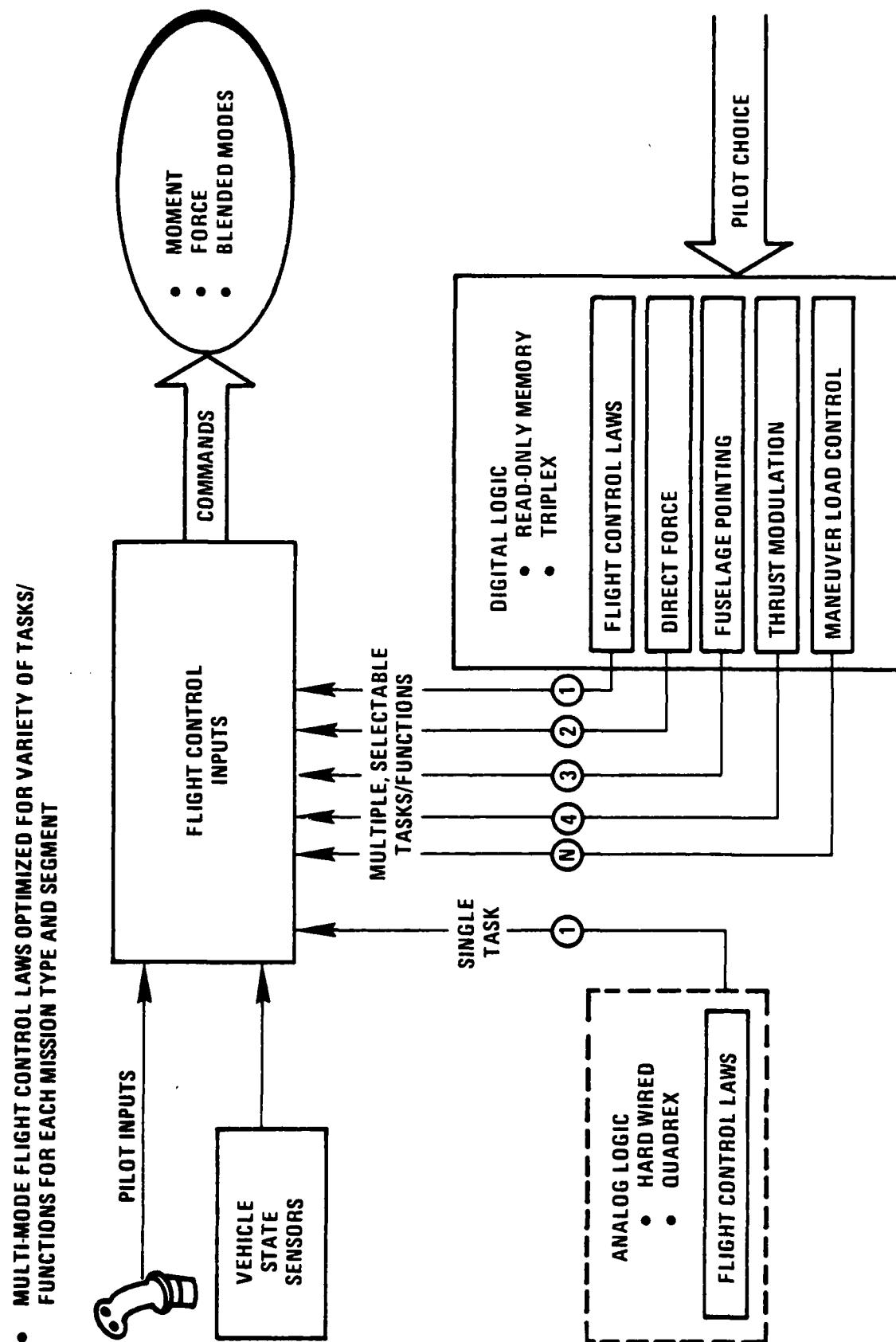


Figure 24. Benefits of a Fly-By-Wire FCS with Digital Processing

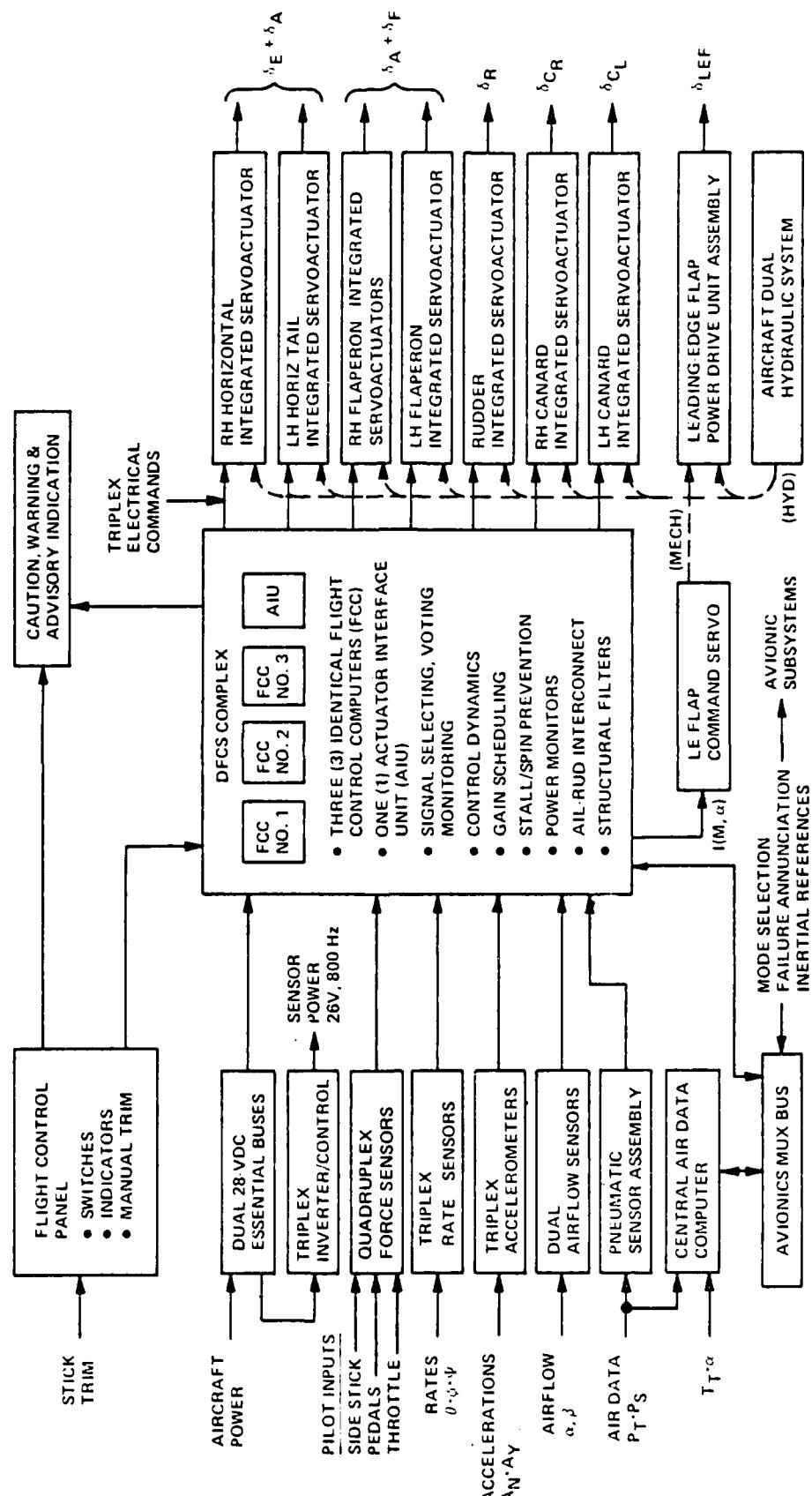


Figure 25. Functional Schematic of the AFTI/F-16 DFCS

of an analog type FBW FCS. Incorporation of the RSS concept has been justified by its contribution to the performance, maneuverability, and range of the F-16. In subsonic flight the horizontal tail (H.T.) is lifting and in supersonic flight the H.T. experiences a down load. RSS permits this down load to be reduced. Reduced H.T. trailing edge up deflection provides the following benefits:

- Increased aerodynamic lift
- Reduced trim drag

These benefits in turn result in:

- Reduced wing area requirement which in turn allows a reduction in structural weight and skin friction drag.
- Reduced fuel requirements for a given range, or increased range for a given amount of fuel.

Three basic systems comprise the FBW type FCC's of the F-16 series aircraft and are listed as follows:

1. Primary Flight Control System (PFCS)

Provides three-axis flight path control through the deflection of the primary control surfaces.

2. Secondary Flight Control System (SFCS)

Provides high lift, aerodynamic braking, and improved maneuvering performance through the deflection of lift and drag modulation devices.

3. Air Data System (ADS)

Provides aerodynamic intelligence for the FCS and other using systems. Sensed aerodynamic parameters are inserted in the flight control law equations for computation by the FCC's.

The optimal integration of these three systems results in FCS's that possess a variety of advanced capabilities and unique features. Those common to both the analog and digital FCS's will be listed initially in Table 3. The additional capabilities and unique features of the AFTI/F-16 DFCS will be listed subsequently in Table 4.

The basic FCS functional characteristics of the operational F-16's with analog FBW systems are presented in Table 5 (obtained from Reference 9) and are categorized into the following areas:

- Primary Control Surfaces
- Secondary Control Surfaces
- Trim
- Pilot's Primary Controls.

The data in Table 5 is also applicable to the AFTI/F-16 DFCS and provides insight as to the pilot controller inputs used to command the required combinations of control surface deflections to create pitch, roll, and yaw moments and to trim the aircraft. However, some of the quantitative

TABLE 3. CAPABILITIES AND FEATURES COMMON TO BOTH THE
CURRENT F16 AND THE AFTI/F-16

- Full FBW FCS provides maximum flexibility for optimizing flying qualities and the flight control task over the entire flight envelop.
- Use of the RSS concept with its many benefits.
- Three-axis Command Augmentation System (CAS) and Stability Augmentation System (SAS) provides precise control and excellent handling qualities.
- Automatic angle-of-attack (AOA) and normal load factor limiting allow the pilot to use maximum capability of the airplane without inadvertent loss of control.
- Aileron-rudder interconnect provides improved high AOA handling qualities.
- Combined programming of the leading and trailing edge flaps provides optimized airfoil camber for high L/D during cruise and maneuvering flight.
- Effective leading edge and trailing edge flaps provide good take-off and landing performance.
- Effective speed brake provides good speed control in maneuvering flight.
- Redundancy of components and signal paths contributes to high probability of mission completion and increased flight safety.
- Pilot's side-stick controller enhances precise control at high load factors.
- Built-in test capability ensures FCS flight readiness with minimal downtime for maintenance actions.

TABLE 4. CAPABILITIES AND FEATURES UNIQUE TO
THE AFTI/F-16

- Six independent degrees of freedom flight control capability over the entire flight envelop.
- The following advanced longitudinal control modes:
 - Direct lift (direct control of A_n)
 - Fuselage pointing (α_1 mode)
 - Vertical translation (α_2 mode)
- The following advanced directional control modes:
 - Direct sideforce (direct control of A_y)
 - Fuselage pointing (β_1 mode)
 - Lateral Translation (β_2 mode)

**TABLE 4. CAPABILITIES AND FEATURES UNIQUE TO
THE AFTI/F-16 - Continued**

- Maneuver enhancement/gust alleviation to provide the following benefits:
 - Automatic direct lift through force stick error signal
 - Quickened pitch response
 - More precise target tracking
 - Reduced aircraft response to random turbulence and gusts
 - Elimination of "g" lag
- Digital type FCC's permit optimal integration of flight, fire, and propulsion control technology advancements.
- Use of in-flight integrity management to improve flight safety.
- Automatic weapon line pointing.
- Reduced redundancy levels to reduce costs, spares, weight, space, and power requirements and retain typical FCS high reliabilities.
- Integrated flight controls and Flight Management Display System (FMDS) to reduce pilot workload and simplify cockpit layout.

values in Table 5 will not be exactly the same as for the AFTI/F-16. The table provides no data on the canards of the AFTI/F-16. The use of the canards to enhance multi-mode capabilities will be discussed later.

1. Primary Flight Control System (PFCS)

The PFSC interfaces directly with the Secondary Flight Control System (SFCS) and the Air Data System (ADS). The interface with the ADS is shown in Figure 26.

Longitudinal or pitch control is achieved through fore-and-aft forces applied to the Side Stick Controller (SSC) in addition to the SSC pitch commands the following signals also influence pitch control:

- Angle-of-attack (α)
- Dynamic pressure
- Ratio of dynamic pressure to static pressure (q_C/P_S)
- Pitch rate
- Differential commands to the horizontal tails for augmenting roll control
- Pitch trim
- Normal acceleration
- Autopilot inputs

TABLE 5. F-16 FLIGHT CONTROL SYSTEM FUNCTIONAL CHARACTERISTICS

PRIMARY CONTROL SURFACES

AXIS	LONGITUDINAL	LATERAL		DIRECTIONAL
Type of Surface	Horizontal Tail	Flaperon	Asymmetrical Horiz Tail	Rudder
Deflection (Max) (1 to Hinge Line)	±25 Deg	+20 to -23 Deg	±5 Deg (Min.) ±10 Deg (Max.)	±30 Deg
Surface Rate Limit (Max, No Load)	60 Deg/Sec	52 Deg/Sec	±60 Deg/Sec	120 Deg/Sec @ 0 H.M.
Stall Hinge Moment (Both Hyd Systems)	14,400 Ft-Lb @ 0 Deg	15,850 Ft-Lb @ 20 Deg	14,400 Ft-Lb @ 0 Deg	4,700 Ft-Lb @ 12 Deg
Stall Hinge Moment (One Hyd System)	7,200 Ft-Lb @ 0 Deg	7,925 Ft-Lb @ 20 Deg	7,200 Ft-Lb @ 0 Deg	2,350 Ft-Lb @ 12 Deg

SECONDARY CONTROL SURFACES

TYPE OF SURFACE	LEADING EDGE FLAP	FLAPERON	SPEED BRAKE
Deflection (Max) (Streamwise)	-2 to +25 Deg	+20 to -23 Deg (T.E. Flap)	0 to 60 Deg
Surface Rate Limit (Max, No Load)	30 Deg/Sec @ 0 H.M.	5 Deg/Sec (flap function)	30 Deg/Sec (Per Surface)
Stall Hinge Moment (Both Hyd Systems)	175,000 In-Lb @ 0 Deg	15,850 Ft-Lb @ 20 Deg	Single System Only
Stall Hinge Moment (One Hyd System)	87,500 In-Lb @ 0 Deg	7,925 Ft-Lb @ 20 Deg	6,065 Ft-Lb/Surface
Command Servo	Dual Position & Rate Electrical Feedback	None	None

TRIM

AXIS	LONGITUDINAL	LATERAL	DIRECTIONAL
Type of Surface	Horizontal Tail	Flaperon and Horiz Tail Variable Ratio	Rudder
Authority	±2.4 G's (Surface Deflection a Function of Variable Gain)	±66.7 Deg/Sec	±12 Deg
Trim Rate	±.90 G's/Sec from Pilot's Controller	±4.5 Deg/Sec/Sec from Pilot's Controller	Set by Rate Pilot Turns Knob on Panel

PILOT'S PRIMARY CONTROLS

AXIS	LONGITUDINAL	LATERAL	DIRECTIONAL
Type	Pilot's Sidestick Controller	Pilot's Sidestick Controller	Rudder Pedals
Displacement	Minimum	Minimum	±0.5 In.
Breakout	±1.75 Lb	±1.0 Lb	±15.0 Lb
Maximum Force	Max Command (+10.86 G's at 40.0 Lb Aft) (-4 G's at 18.44 Lb Fwd)	Max Roll Command (308 Deg/Sec at 17.0 Lb)	Max Surface at 110 Lb

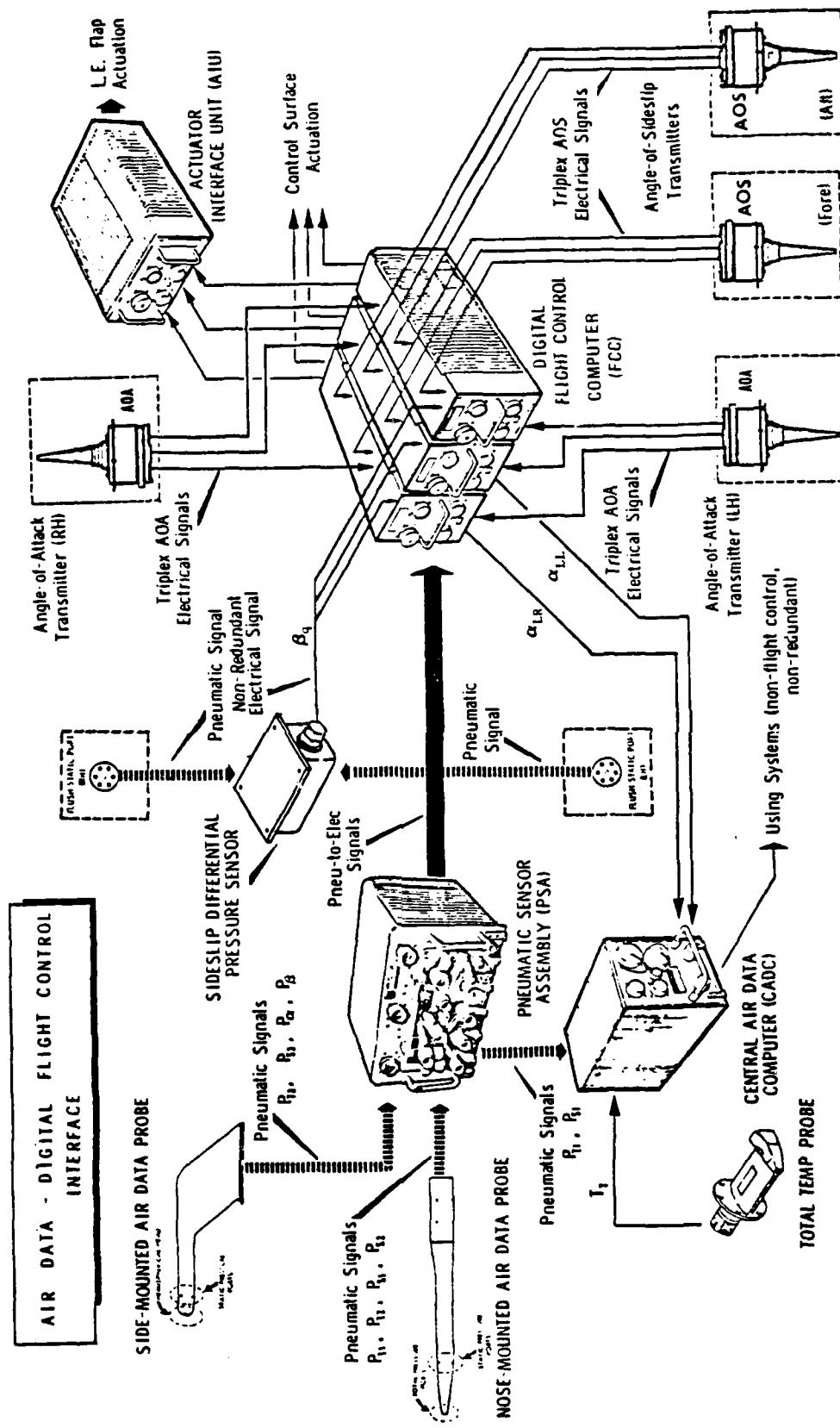


Figure 26. Air Data System Interface with the DFCS

The pilot applies pitch trim by displacing the four-way trim button on the grip of the side stick. Fore and aft displacements of the trim button drive an electrical gear-head motor that is located in the manual trim panel and connected to a stack of three potentiometers and a thumb-operated pitch trim wheel. Triplex trim signals are generated, summed with other pitch command signals, gain-adjusted, and sent to both the left and right horizontal tail integrated servoactuators (ISA).

Lateral or roll control is achieved through lateral forces applied to the SSC. The roll axis employs both CAS and SAS as an aid to achieving precise flight path control. Roll CAS is a roll-rate-referenced command system capable of minimizing the roll response per unit of lateral side stick force. Roll command at any instant is a function of the following:

- Lateral side-stick force
- Roll rate
- Roll trim
- Autopilot commands

Both flaperon and differential horizontal tail deflections are used for roll control to assure that adequate roll control power is available throughout the flight envelope.

Roll trim is achieved in a manner similar to pitch trim. Roll trim is accomplished by lateral displacement of the four-way trim button on the grip of the side stick controller. These lateral trim commands drive an electric gear-head motor in the manual trim panel. Triplex trim signals are generated, summed with other signals and gain-adjusted to provide a variable trim distribution to the flaperons and the horizontal tails in accordance with the horizontal tail-flaperons ratio schedule.

Directional or yaw control is achieved by the use of the rudder pedals. The yaw axis employs basic stability augmentation for yaw damping and automatic turn coordination through the use of lateral acceleration referenced command augmentation. A gain adjusted aileron-rudder interconnect (ARI) and a roll rate-to-rudder cross feed minimize roll-yaw coupling, extend the AOA for departure boundary, improve tracking, increase spin resistance, and provide improved turn coordination. Rudder commands at any instant are a function of the following parameters:

- Rudder pedal force
- Yaw trim
- Yaw rate
- Lateral acceleration

The pilot trims the aircraft in yaw by rotating the yaw trim knob, on the manual trim panel, which is connected to a stack of three potentiometers which generate triplex yaw trim signals. These signals are summed with other signals in the FCC's and sent to the rudder ISA.

2. The Secondary Flight Control System (SFCS)

The Secondary Flight Control System (SFCS) consists of the following elements:

- Full-span leading-edge flap (LEF).
- Flaperons functioning as trailing edge flaps (TEF).

- Speed brakes located on each side of the aft fuselage.
- Canards functioning as speed brakes by deflecting the leading edges of each inward (toe-in).

The LEF is scheduled as a function of Mach number and AOA with the scheduling being based on performance improvement and buffet reduction rather than on stability and control requirements. The LEF provides the following benefits:

- Increased lift for take off and landing.
- Automatic and near optimal wing camber as a function of flight condition thereby providing improved turning capability and increased effectiveness of the vertical fin at high AOA.

TEF flap and roll commands are summed in the FCC's to produce a desired flaperon deflection. In the F-16 FBW systems no additional components are required to accomplish the TEF function except for the electronic components required to accept the flap command in the FCC's. The absence of a flap command to the FCC's results in a zero flap command to the flaperons.

The speed brake consists of two pairs of clamshell surfaces located adjacent to the engine nozzle and inboard of the horizontal tail surfaces in the trailing edge of the left- and right-hand horizontal tail shelf structure. See Figure 27 obtained from Reference 9. Each speed brake consists of the following:

- A double acting hydraulic actuator.
- An upper door with linkage.
- A lower door with linkage.
- Three hinged seals that are activated by the motion of the speed brake.

Figure 27 shows in detail the mechanization of the speed brakes, location, and maximum available deflection.

AFTI/F-16 AIR DATA SYSTEM (ADS)

The configuration of the AFTI/F-16 Air Data System is shown in Figure 26. This configuration is basically the same as that of the F-16 with the exception of the addition of two sideslip probes, which are identical to the AOA probes, and repositioning of the right-side air data probe to eliminate canard interference. The Electronic Component Assembly (ECA) used in the F-16 ADS is not required in the AFTI/F-16 ADS because its functions can be performed in the digital FCC's.

The AFTI/F-16 ADS is comprised of the following units:

- Nose mounted air data probe.
- Fuselage side-mounted air data probe.
- Two flush fuselage-mounted static pressure ports.
- Two radome mounted AOA transmitters.
- Nacelle mounted total temperature probe.
- Pneumatic Sensor Assembly (PSA)

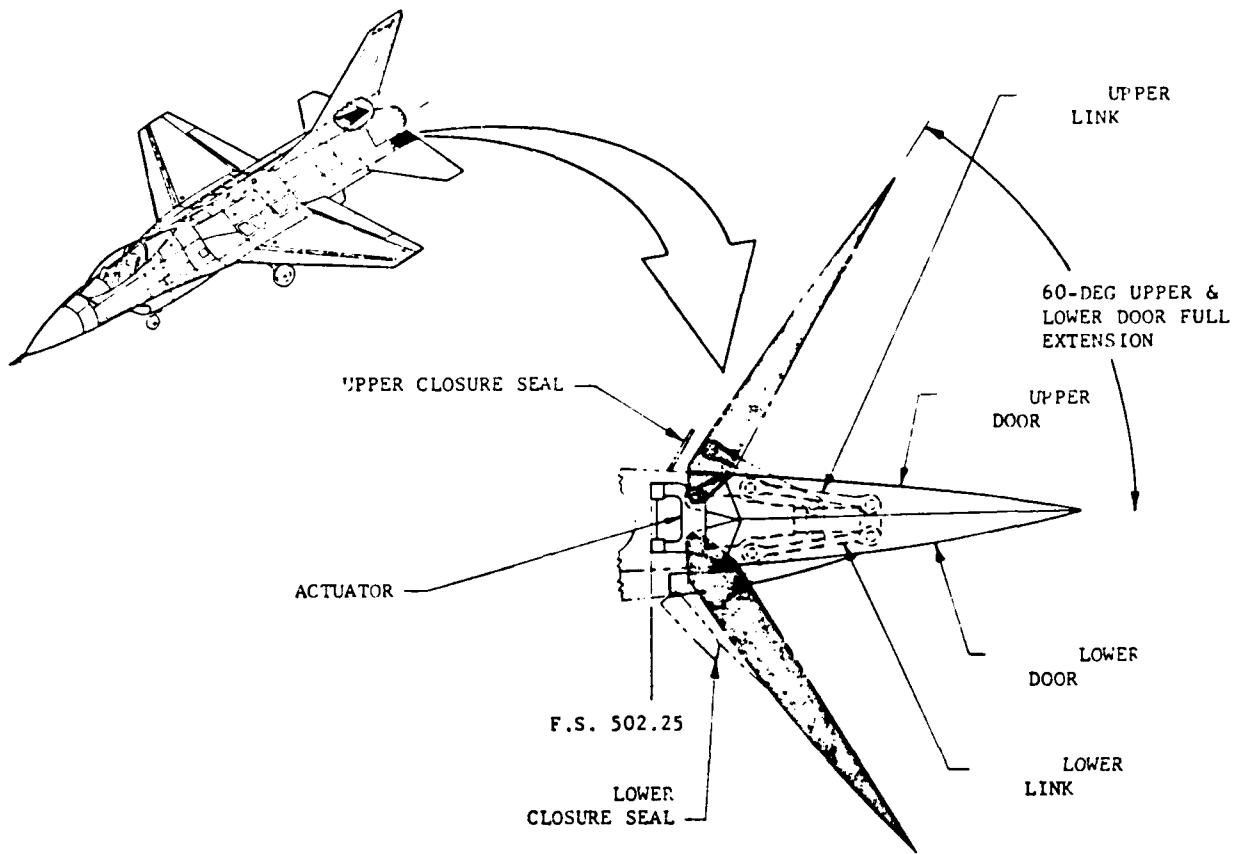


Figure 27. Mechanization of the Speedbrake

- Central Air Data Computer (CADC).
- Sideslip differential pressure sensor.
- Two Angle-of-Sideslip (AOS) transmitters.

Three independent sources of total pressure (P_T), static pressure (P_S), AOA, and AOS are sensed and converted to electrical signals. Middle-value selection of the air data parameters are employed to provide triplex identical signals to be inputted to the FCC's for control law computation, gain scheduling, and LEF computation.

1. Air Data Probes -

The nose mounted in data probe provides dual sources of static (P_S1 and P_S2) and total (P_T1 and P_T2) pressures. Aerodynamic compensation is provided to minimize the static pressure source position error. The multi purpose fuselage air data probe (also referred

to as the side-mounted probe) provides a single source of static and total pressure and AOA. Aerodynamic compensation is provided to minimize the static pressure source position error. This multi-purpose probe provides five pneumatic outputs (P_{T3} , P_{S3} , $P_{\alpha1}$, $P_{\alpha2}$, and P_{β}) as inputs to the Pneumatic Sensor Assembly (PSA) where pressure transducers convert pneumatic signals to electrical analog signals. The method of sensing AOA is the same as that described for the multipurpose probe shown in Figure 3. The AOS ports are pneumatically connected together to provide a pseudo-static pressure (P_{β}) reference.

A nacelle mounted total temperature probe provides the Central Air Data Computer (CADC) with a total temperature (T_T) signal (analog). This signal is required by the CADC for true airspeed and air density ratio computations. The probe is installed on the left side of the engine nacelle so that the intake of the total temperature probe is located outside of any boundary layer and not in the wake of any upstream portion of the aircraft.

2. Static Pressure Ports

Two flush-mounted static pressure ports provide inputs to a Sideslip Differential Pressure Sensor for AOS measurement. This measurement is used to compensate the third AOA source (Side-Mounted Air Data Probe) error. The flush ports are located diametrically opposite to each other and forward of the cockpit.

3. Sideslip Differential Pressure Sensor

This sensor is installed remotely to the PSA to provide an AOS function to the digital Flight Control Computers (FCC). The sensor output is a function of AOS and dynamic pressure.

4. Angle-of-Attack (AOA) Transmitters

Two conical air-flow-detector type AOA transmitters are mounted diametrically opposite on each side of the radome. Each transmitter contains three identical rotary voltage differential transformer (RVDT) outputs, one output to each of the FCC's (see Figure 26). These transmitters convert conical probe rotation (a function of airflow striking probe slots) to electrical analog signals.

5. Angle-of-Sideslip (AOS) Transmitters

These units are AOA transmitters mounted on the underside of the fuselage in a manner to sense AOS. Their triplex outputs are sent to the FCC's (see Figure 26).

6. Pneumatic Sensor Assembly (PSA)

The PSA contains eight pressure sensors that convert pneumatic inputs from the nose and fuselage mounted air data probes into electrical signals. The differential pressure type AOA signal is normalized to dynamic pressure at the hemispherical head of the side mounted air data probe. This dynamic pressure is provided by subtracting the average AOS pressures (pseudo-static pressure) from the total pressure at the hemispherical head. Normalization (division) is achieved by use of a multiplier as a feedback element in an operational amplifier circuit. The PSA supplies triplex static and dynamic pressure signals to the FCC's.

7. Central Air Data Computer (CADC)

The CADC is a digital computer which accepts inputs from the nose-mounted air data probe, total temperature probe, and the two AOA transmitters. The CADC consists of the following six basic functional sections:

1. Power Supply
2. Pressure Sensors (2)
3. Input Data Converter (IDC)
4. Digital Processor (DP)
5. Output Data Converter (ODC)
6. Chassis Assembly

The values of P_{T_1} and P_{S_1} are inputted into the Input Data Converter (IDC) via two pressure sensors and the barometric, AOA and total temperature signals are inputted directly to the multiplexer and A/D converter section of the IDC. The IDC converts the data from these inputs to digital form and transmits them to the digital processor (DP) section for processing. The input data is corrected for source errors in the computer. The required CADC outputs are calculated, formatted, and transferred to the output data converter (ODC) section. The ODC converts the data to the appropriate signal form (digital, analog, synchro, discrete) and distributes the following air data parameters to the using systems:

- Pressure altitude (H_p) (referenced to 29.92 in Hg)
- Pressure altitude (H_{PBC}) (referenced to altimeter barometric setting)
- Pressure altitude rate (\dot{H}_p)
- Pressure ratio (P_S/P_{SL})
- True temperature (T_{FAT})
- Calibrated airspeed (V_C)
- True airspeed (V_T)
- True angle-of-attack (α)
- Air density ratio (σ)

8. Air Data System Redundancy Management

All air data parameters essential for flight control law computation are inputted to the FCC's to a triplex level of redundancy. Redundancy management (failure detection, isolation, and reconfiguration) of these inputs are performed by the FCC's. Initial failures are detected by middle value voting techniques and the failed input is isolated thereby permitting single-failure operational capability. After the occurrence of a second like failure it is not possible to isolate the failed input by using voting techniques and the data from two sensors is ignored. The FCC's will then employ a backup set of flight control laws that provide alternative control laws that provide at least Level 2 flying qualities with a goal of level 1.

After the failure of a second AOA sensor the AOA feedback will be removed and not used as an aid in the computation of control laws. In the cases of dual failures of the other air data sensors fixed gains will be used in control law computation.

FLIGHT CONTROL COCKPIT CONTROLS

The AFTI/F-16 flight control cockpit controllers consist of a right side mounted Side-Stick Controller (SSC) Assembly, a Rudder Pedal Assembly, an FCS Control Panel, and a Rotatable Throttle Controller Grip for pitch pointing control.

1. Side-Stick Controller (SSC) Assembly

The SSC is equipped with physically and electrically isolated transducers so that applied forces (resulting in only small total displacements) to the stick grip will generate quadrex electrical command signals. These signals are summed with other signals, gain-adjusted in the FCC's, and transmitted to the primary control surface integrated servo-actuators. Linear variable differential transformer (LVDT) type transducers are used in these applications.

The pilot achieves longitudinal pitch control through fore-and-aft forces applied to the minimum-deflection type force-sensing controller. In a similar manner, the pilot achieves lateral or roll control through lateral forces applied to the SSC. Figure 28 shows the installation and assembly of the SSC for the F-16 and was obtained from Reference 9.

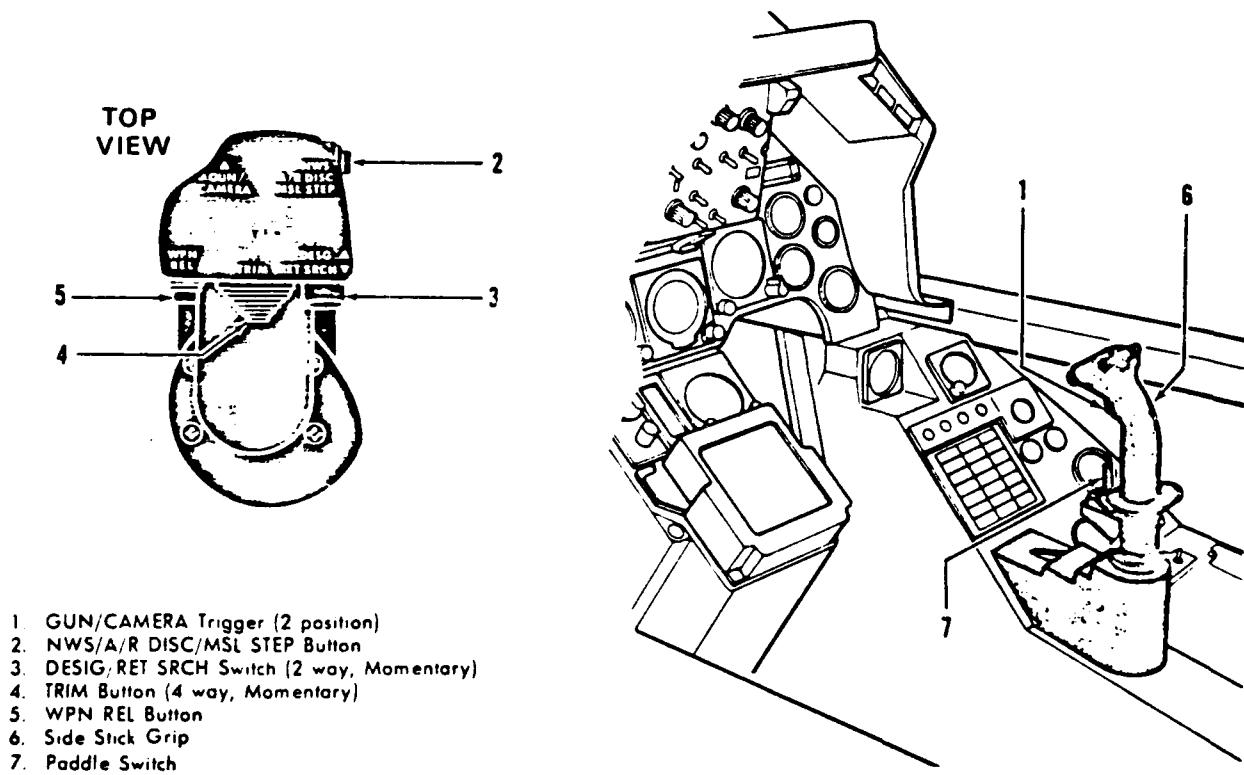


Figure 28. Installation and Assembly of the Side-Stick Controller (SSC) for the F-16

The AFTI/F-16 SSC grip is the same as the F-16 grip except that two switch housings have been added, one 4-way slide switch is located where the F-16 grip has a 2-way slide switch, and since the autopilot disconnect is not required, the space is used for the Integrated Flight and Fire Control (IFFC) switch.

The switch housings were added to provide room for three new redundant switches. The Independent Backup Unit (IBU) and Control Configured Vehicle (CCV) switches are located in the top housing. A flip-over paddle was added to the top housing to provide switch actuation for the CCV switch. The IFFC switch is located in the lower housing on the aft side of the stick and is operated through the paddle located at the lower forward side of the stick.

The new 4-way slide switch was added to provide the Helmet Mounted Sight (HMS) switching feature. The switch is located in the same position as the 2-way DESIG-RET SRCH is located on the F-16 grip. The HMS position is on the inboard side. A space position is on the outboard side.

The trigger switch is the same as for the F-16 grip except the name has been changed from CAMERA/GUN to RCD-LASER/GUN. The descriptions of the AFTI/F-16 grip was obtained from reference 10. Figure 29 shows the AFTI/F-16 Side-Stick Controller (SSC).

The AFTI/F-16 SSC will be essentially two-fail operative. Quadrex force transducers in both the pitch and roll axes are used as an aid in detecting and isolating single and dual failed force transducers. Figure 30 is a cross sectional view of the transducer portion of the SSC showing the locations of the pitch and roll LVDT's, four per control axis.

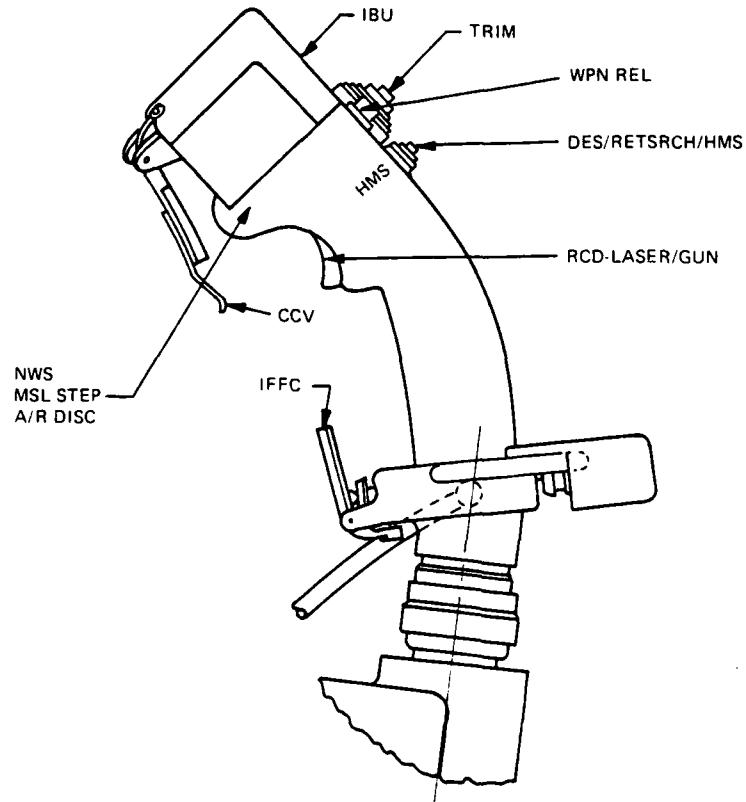


Figure 29. AFTI/F-16 Side Stick Controller (SSC)

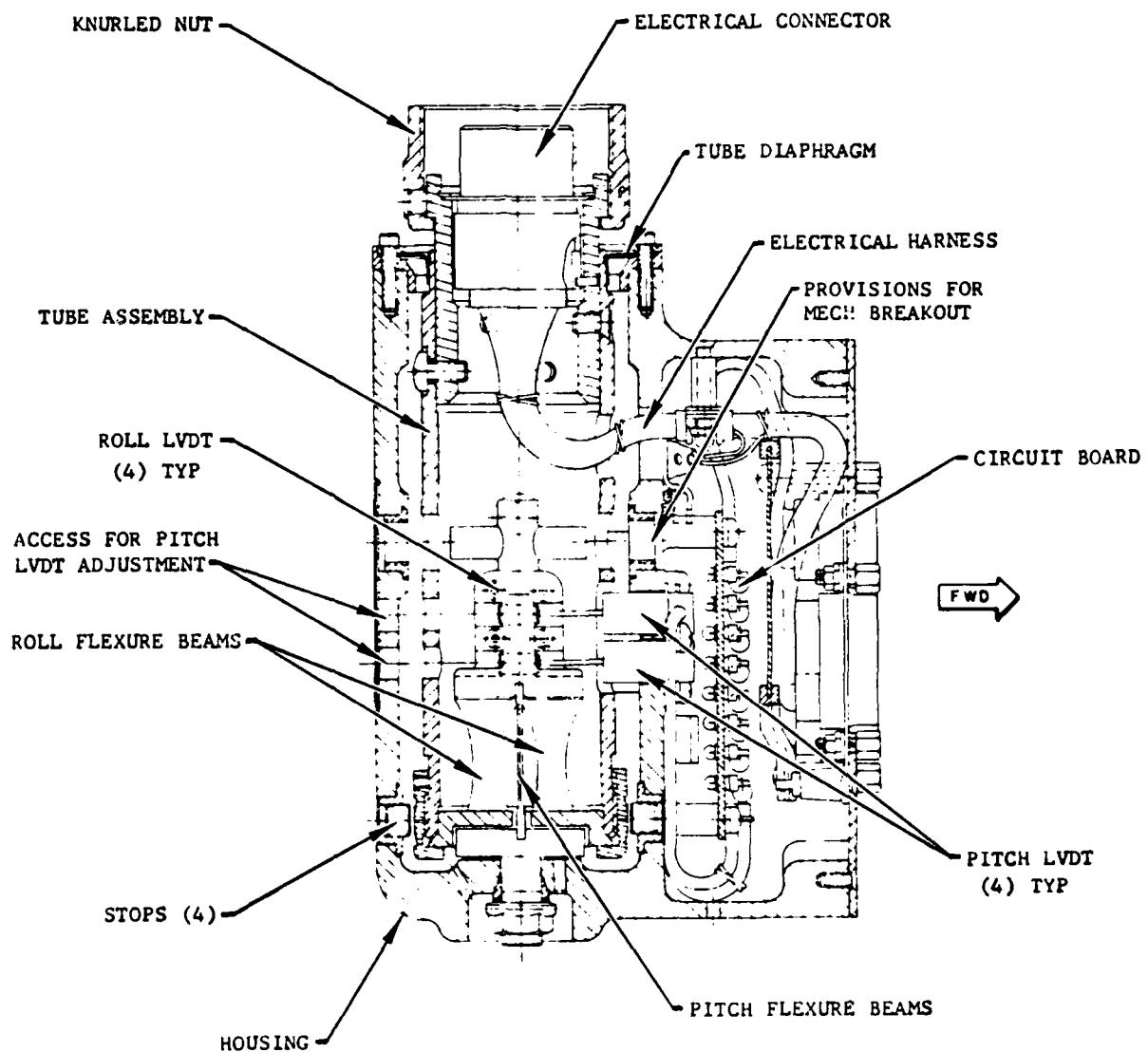


Figure 30. Cross-Sectional View of Transducer Portion of the Side-Stick Controller

2. Rudder Pedal Assembly

The rudder pedal assembly for the AFTI/F-16 is the same as for the F-16. This assembly consists of a minimum-deflection force-sensing pair of conventional rudder pedals. The pilot achieves directional control through translation of either pedal (approximately ± 0.5 inch) to generate quadrex electrical signals using an LVDT type transducer. These signals are summed with other gain-adjusted signals in the FCC's and transmitted to the rudder ISA. Rotation of either pedal generates dual electrical brake command signals from two LVDT type transducers. The rudder pedal assembly is also used for nose wheel steering.

The rudder pedals are used for yaw pointing of the aircraft when the FCS is configured to operate in one of the pilot selected advanced control modes. Figure 31 illustrates the configuration of the rudder pedal assembly.

The AFTI/F-16 Rudder Pedal Assembly is also two-fail operative. Quadrex force sensors are used as an aid in isolating single and dual failed force transducers.

3. Flight Control Panel

The Flight Control Panel Assembly is located on the pilot's left hand side as shown in Figure 32. It provides a redundant or backup set of trim and reset switches. The components on the panel are:

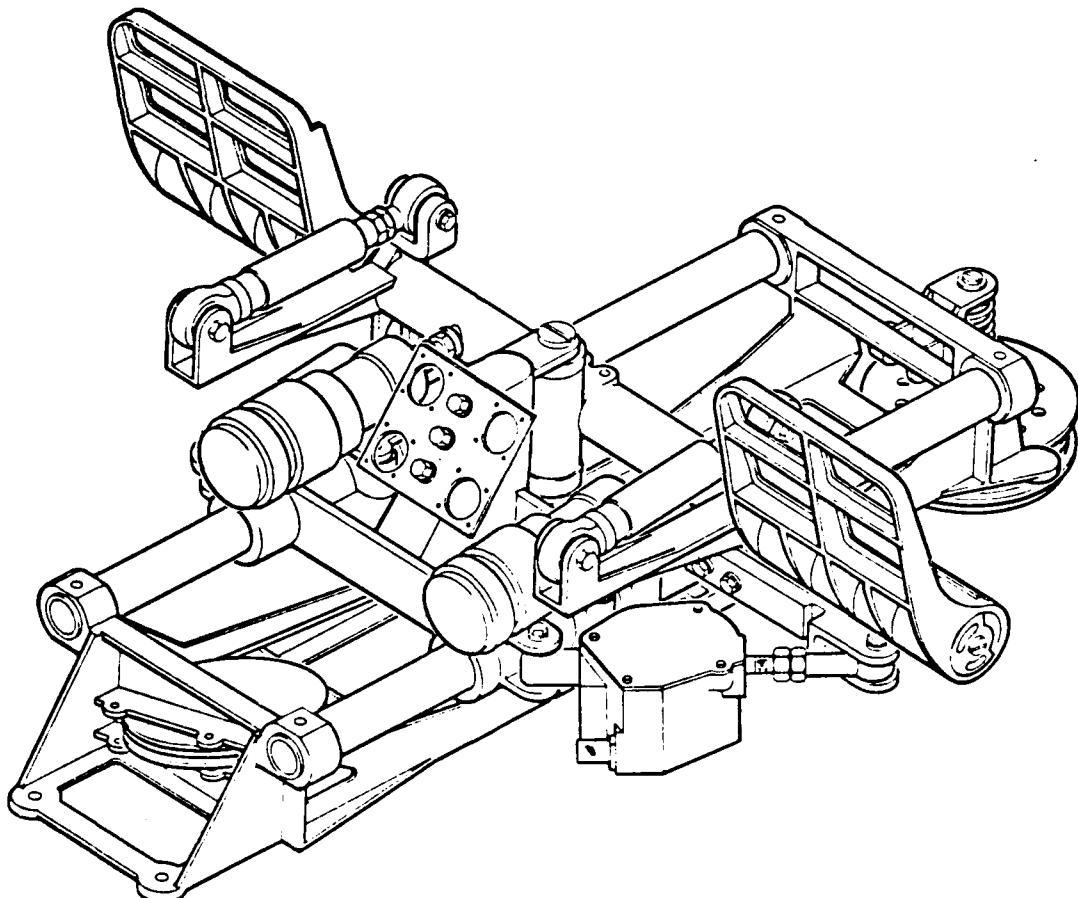


Figure 31. Rudder Pedal Assembly

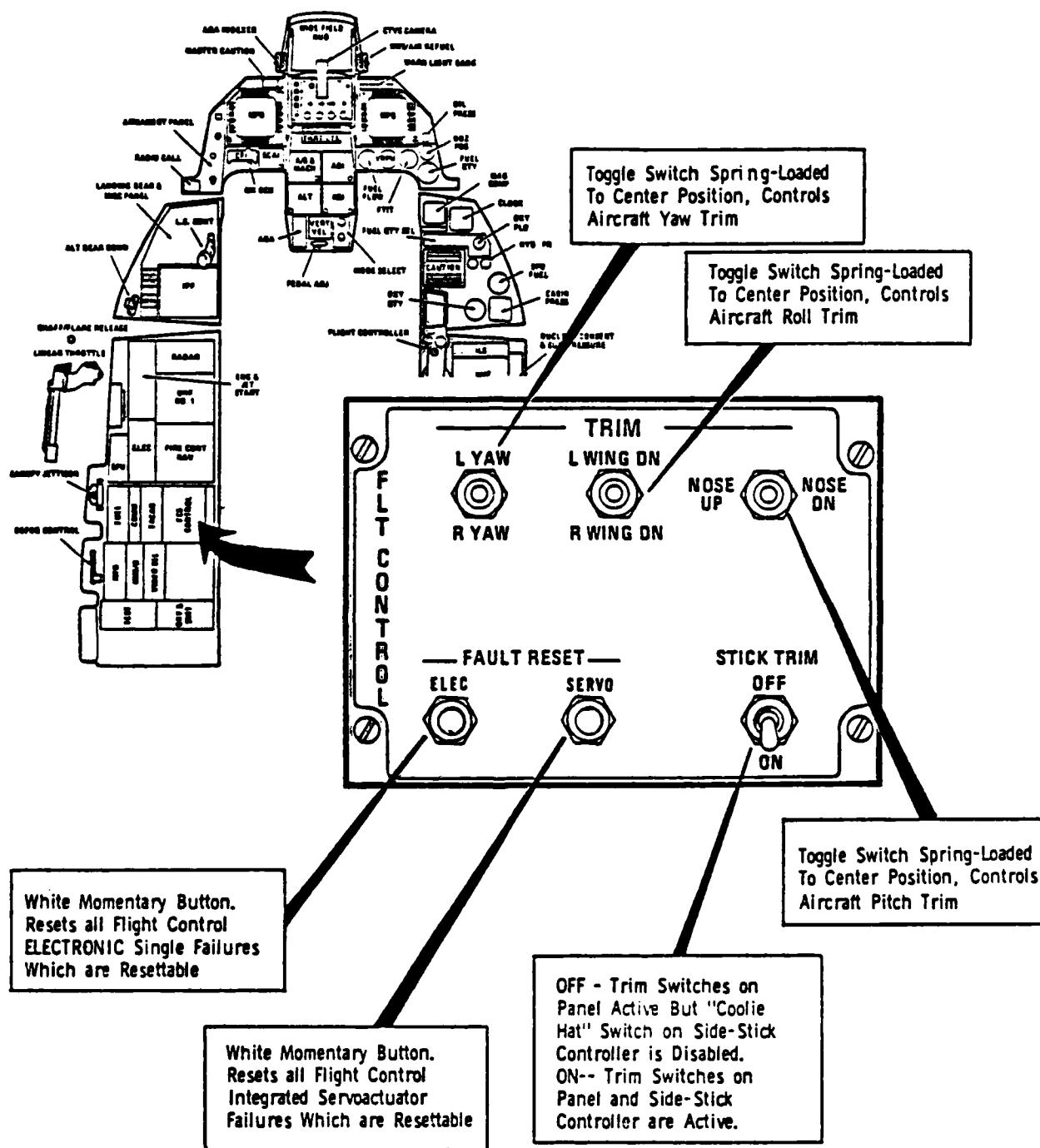


Figure 32. Flight Control Panel Installation

- Redundant Trim Switches (3 position toggle)
 - 1. L YAW
R YAW
 - 2. L WING DN
R WING DN
 - 3. NOSE UP
NOSE DN
- Fault Reset Switches (pushbutton)
 - 1. Electronic
 - 2. Servo
- Redundant Stick Trim Switch
(2 position toggle: OFF-ON)

When the Stick Trim Switch is positioned to "ON", trim functions can be controlled from the stick trim button on the FSC or from the Flight Control Panel. The "OFF" position of the Stick Trim Switch deactivates the SSC trim button so that trim can only be controlled through the Flight Control Panel trim switches. The backup trims are required because the SSC trims are nonredundant. The panel trim functions are implemented as triply redundant discrete inputs to the FCC's where the trim integration function resides in software. The reset buttons are included on the panel to permit "blind" failure resets for those rare instances when both Multi-Purpose Displays (MPD) are not operational. The panel resets operate essentially the same as the MPD resets in the way they reset a first or second like failure.

The term second like failure refers to two independent failures which have been detected by the same monitor and does not necessarily imply that the two failures were physically identical. For example, given that the first failure was branch A pitch rate gyro, then a second like failure would include a pitch rate input demodulator in branch B or C, an A/D converter failure in B or C, an 800 Hz power supply in B or C as well as an identical pitch rate gyro failure in branches B or C.

There are no limitations as to which failures may be reset or the number of times a reset request may be repeated. The FCS Operational Flight Plan (OFP) presents a second failure from being reset back to the no failed state and creating the chance for an apparent simultaneous dual like failure situation with potential loss of control. The price paid for this added protection is that the first failure must be latched in permanently whenever a second like failure occurs.

4. Rotatable Throttle Controller Grip

Rotation of the Rotatable Throttle Controller Grip provides the controlling inputs of the following CCV modes upon pilot selection:

- Pitch Pointing
- Vertical Translation
- Direct Lift

The entire Throttle Assembly as shown in Figure 33, is located in the cockpit above the left console and is operated by conventional fore and aft motion. The throttle grip rides on a spline shaft on recirculating ball bearing with a total travel of 8.12 inches. Each end of the spline shaft is mounted to aircraft structure with small brackets. The output of the throttle linkage is connected to an enclosed push-pull control to transmit motion to the engine mounted fuel control.

5. Interface With the Flight Control Computers (FCC)

The redundant analog and discrete inputs from the cockpit controllers are supplied to each FCC with the required cross-strap for redundant data selection implemented by a serial digital data exchange between the FCC's. Analog/Digital (A/D) conversion of the inputted cockpit controller signals takes place within the FCC's.

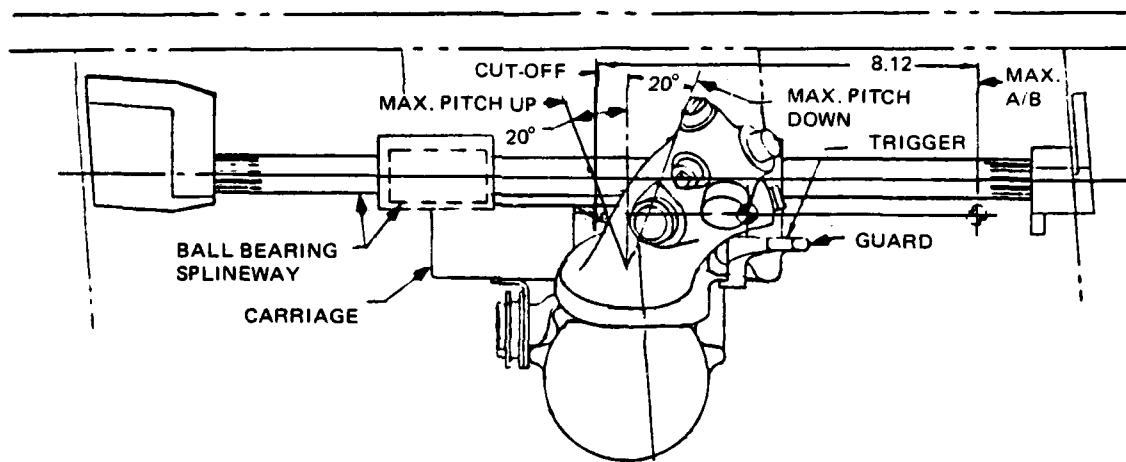


Figure 33. Throttle Assembly

FLIGHT CONTROL INERTIAL SENSORS

The AFTI/F-16 Primary FCS inertial sensors are contained in a Rate Gyro Assembly and an Accelerometer Assembly which are identical to those used in the production versions of the F-16. These are quadrex units designed for use in the quadrex analog type FBW system for the F-16. For use in the AFTI/F-16 only 3 of each of the 4 like sensors will be interfaced with the triplex DFCS.

Dual like inertial sensor failures will not produce a loss-of-control condition. The second like failure will be detected by cross channel monitoring, and a "control law reconfiguration" scheme will be used to maintain safe flight.

1. Rate Gyro Assemblies

Three Rate Gyro Assemblies (one each for pitch, roll, and yaw) consist of four rate gyros each to provide redundant signals as a function of body rates about their respective axes. For the AFTI/F-16 one rate gyro in each assembly will not be interfaced with the triplex DFCS. Figure 34 shows the installation of the Rate Gyro Assemblies. The Rate Gyro Assemblies are identical for each of the three locations and do not require separate stocking for three different applications. Spring-balanced, floated rate gyros are used; each rate gyro contains a permanent magnet torquer for self testing and a monitor of spin-motor lead-phase voltage characteristics. Each rate gyro subassembly contains a phase-splitting capacitor potted in the head pin end), a cable, and external connector.

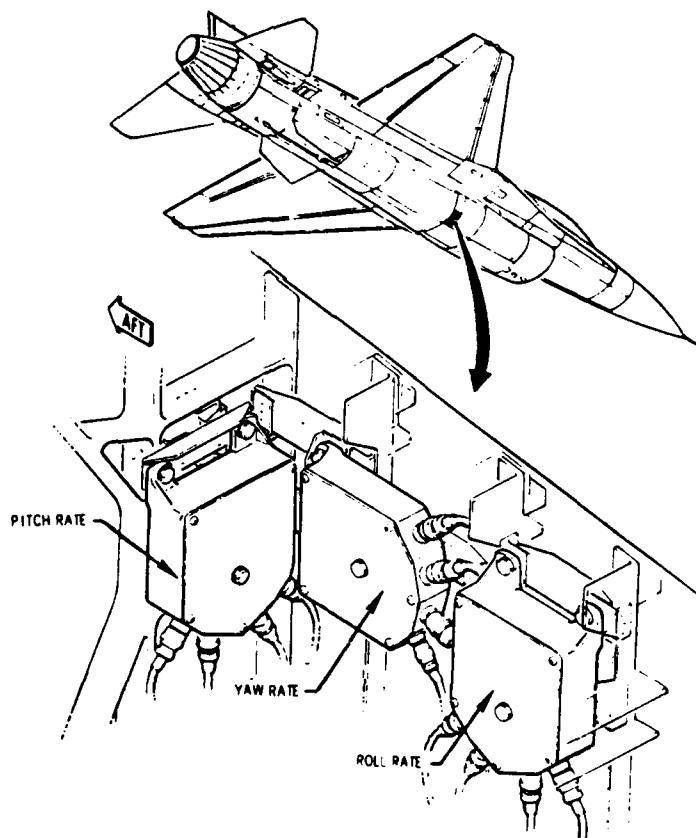


Figure 34. Installation of the Rate Gyro Assemblies

2. Accelerometer Assembly

The Accelerometer Assembly consists of eight identical torque-balanced accelerometers fastened to a common mounting base and enclosed with an aluminum dust cover. Each accelerometer is entirely self-contained, analog closed-loop, sensing transducer having all solid-state silicon electronics. Four of the accelerometers are aligned within the assembly to sense acceleration along the normal axis and four for sensing lateral acceleration. For application to the AFTI/F-16 one of the lateral and of the normal accelerometers will not be interfaced with the triplex DFCS. The installation of the Accelerometer assembly in the aircraft is shown in Figure 35. The location of the installation is consistent with requirements to yield minimum body-bending effects and maximum aerodynamic stability. The physical center of the Accelerometer Assembly is located forward of the aircraft C.G. as near as practicable to the center of percussion.

For self-test purposes, an external electrical command is summed linearly in the torquing coil of the accelerometer. The accelerometer mounting flange, located near the C.G. of the assembly, contains four threaded mounting holes. Each of the four external connectors is equipped with two split cables and end connectors. One split cable is connected to a normal accelerometer; the other, to a lateral accelerometer.

3. Interface with the Flight Control Computers

Figure 36 shows the method of interfacing or inputting the analog outputs of the flight control inertial sensors into the triplex FCC complex. The output of each sensor is directly inputted into one of the FCC's. This input is transmitted to the other two FCC's via the intercomputer data link. All FCC's know the current inputted value from each sensor and each FCC is capable of performing the failure detection, isolation, and reconfiguration task.

4. Redundancy Management and Reconfiguration Technique

Failure detection and isolation of the inputted inertial sensor data is performed at the Input Voting/Monitoring Plane (see Figure 36) which is in the software programmed into the triplex FCC complex. Redundant inertial sensor inputs are supplied to each FCC with the redundant data selection implemented by a serial digital data exchange between the FCC's. First and second like sensor failures are detected by cross-channel monitoring techniques. After a sensor failure the average values of the remaining two sensors are used by the FCC's. After a second like sensor failure no effort is made to determine which of the two remaining sensors is failed and the remaining inputs of the particular inertial quantity being sensed is ignored. The flight control laws which normally utilized the values of the sensed inertial parameter are reconfigured.

The AFTI/F-16 is statically unstable in subsonic flight and must employ artificial means to restore pitch static stability. Control laws employed to artificially restore static stability require aircraft pitch rate as an input. To avoid aircraft pitch divergence and subsequent loss of control after the loss of two pitch rate sensors a backup technique of estimating pitch rate (\dot{q}) is utilized. These estimated values are inputted into a set of backup control laws. Estimation of pitch rate is based on AOA and horizontal tail position.

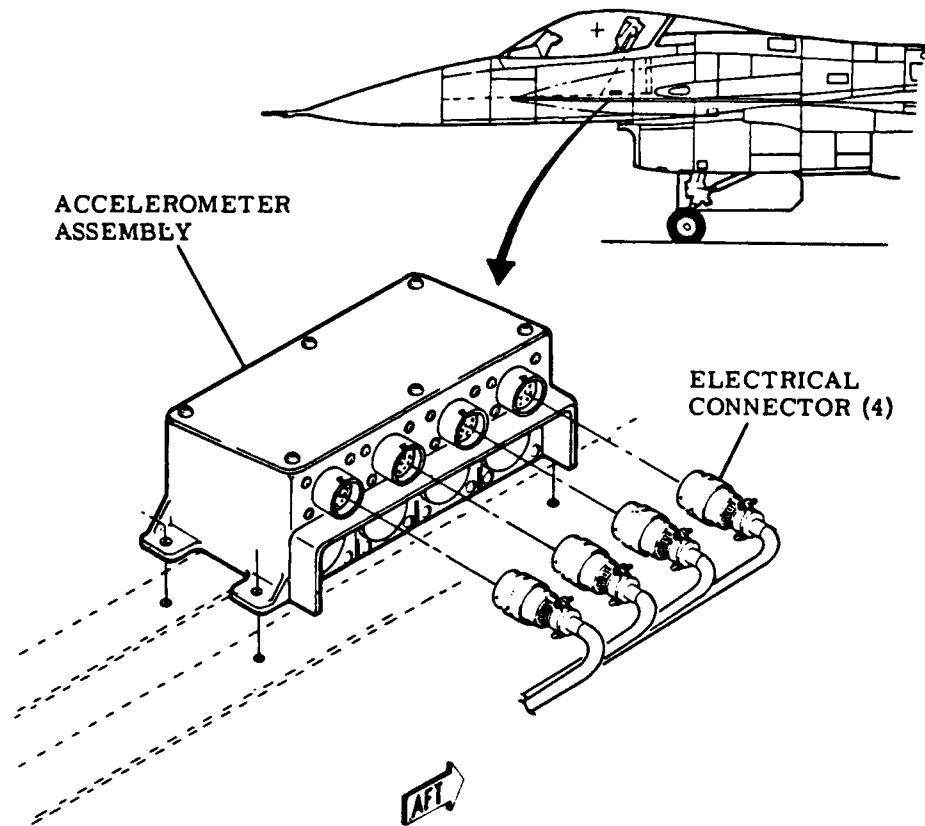


Figure 35. Installation of Accelerometer Assembly

In case of dual loss of like inertial sensors that are only used to enhance performance the reconfigured control law is the non-reconfigured control law with the affected feedback removed. The intent is to provide alternative control laws that provide at least Level 2 flying qualities with a goal of Level 1. Inertial sensor reconfiguration strategy is summarized below

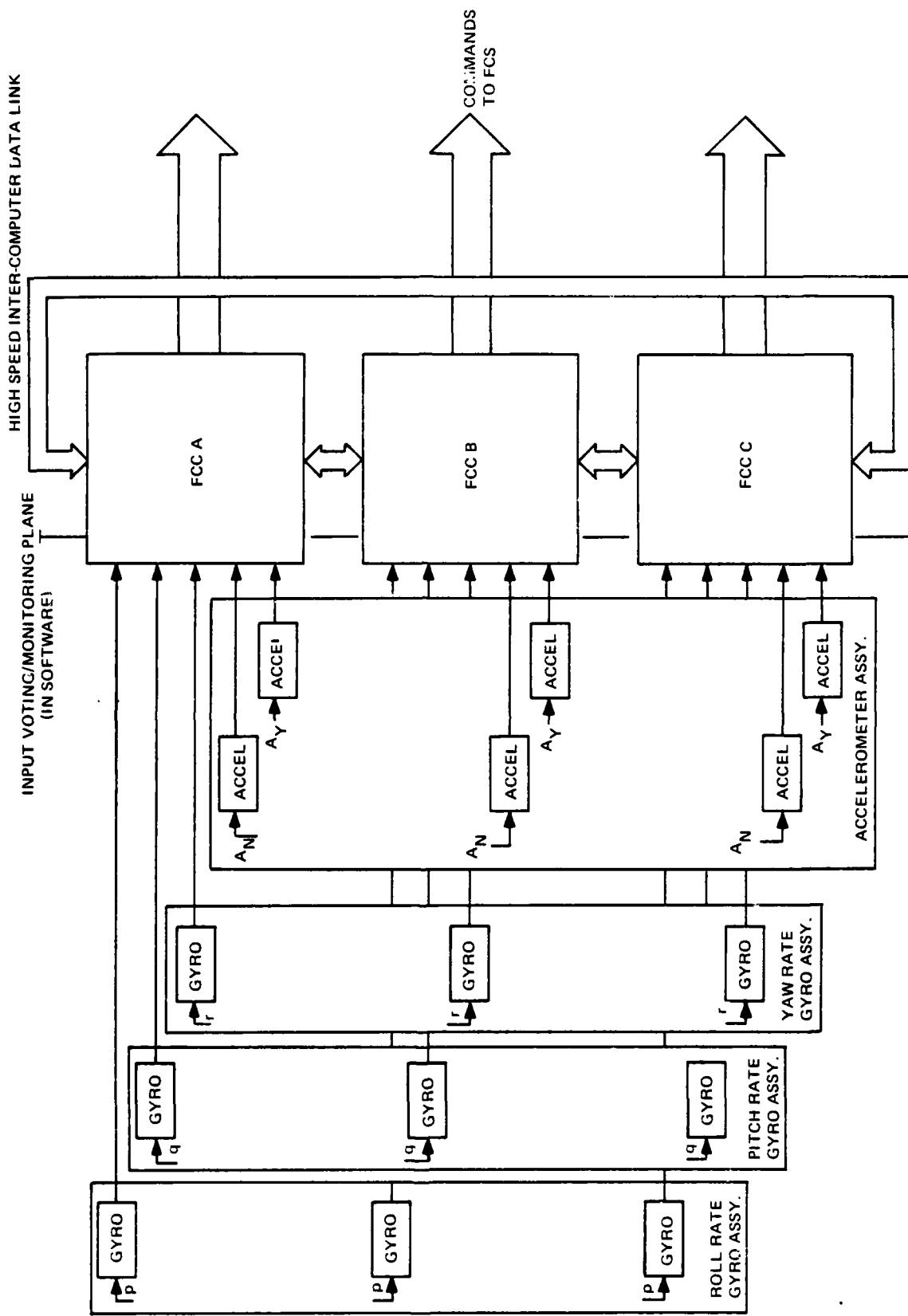


Figure 36. Interface of the Flight Control Inertial Sensors with the Flight Control Computers

SENSORS	RECONFIGURATION STRATEGY AFTER SECOND FAIL MISCOMPARE
• Pitch Rate Gyro	• Estimation of q is based on AOA and horizontal tail position
• Normal Accelerometer	• Reconfiguration to pitch rate command system
• Roll Rate Gyro	• Remove feedback and adjust SSC gain
• Lateral Accelerometer	• Remove feedback
• Yaw Rate Gyro	• Remove A_y and ψ feedbacks, use only the product of $p\alpha$ feedback to rudder

DIGITAL FLIGHT CONTROL COMPUTER (FCC) COMPLEX

The digital FCC Complex consists of three Flight Control Computers (FCC), an Actuator Interface Unit (AIU) and four Inverter Control Assemblies (ICA) which provide AC power for the sensors. The FCC's and AIU were designed and fabricated by the Bendix Corporation. The FCC Complex provides the performance capability required of the FCS for the six DOF AFTI/F-16 aircraft during both ground and airborne operations. The FCC's are advanced third generation Bendix Model BDX-930 processors incorporating innovative redundancy management concepts. The triplex FCC Complex is fully operational after a first FCC failure. The probability of maintaining Operational State 1 performance after a second FCC failure is greater than 95 percent. The three FCC's are physically and functionally identical and contain identical software.

The FCC's provide the computational power required to implement the AFTI/F-16 airframe performance requirements. Signals from airframe sensors and indicators, and electrical power are supplied to the FCC's to permit the required Stability Augmentation System/Command Augmentation System (SAS/CAS) computations. Also supplied to the FCC's are inputs from the pilot's controllers and, through the airplane multiplex bus, interactive mode selection and failure annunciation and recovery. The AFTI/F-16 is controlled by the FCC's and the AIU through appropriate commands to the flight control surfaces.

1. Interface Between FCC Complex and the FCS

The salient interface signals between the FCC Complex and the FCS are shown in Figure 25, most of which being of the analog type. Input filtering is provided on all analog inputs for high frequency noise suppression, and the I/O is protected from EMI and lightning induced electrical transients.

The FCC Complex supplies certain instrumentation output signals, such as angle of attack and sideslip, pitch, roll, and yaw rate, stick and rudder pedal forces, and FCC and AIU temperature. All of these signals are protectively buffered so that a failure external to the FCC Complex will not cause an internal malfunction.

The FCC Complex interfaces with the avionics system via a serial-digital, dual-redundant, time-division multiplex data bus as shown in Figure 37. The FCC Complex responds as a remote terminal to commands received on this bus. The FCC Complex is designed to provide a bus complement of 32 subaddresses of 32 words each so that up to 1000 words of bus traffic is permitted. A failure of an FCC will cause its bus transmitter to stop, and this action will force the bus controller to transfer communication to the second bus and another FCC. The bus interface is designed so that no failure external to the FCC Complex will cause an internal malfunction.

2. The Flight Control Computer (FCC)

A layout/outline drawing of the Bendix BDX-930 FCC is shown in Figure 38. The FCC contains a computer unit that contains a 16-bit paralleled two's-complement binary arithmetic unit and sufficient memory to perform the AFTI/F-16 FCS functions. The BDX-930 FCC is faster, smaller, and consumes less power than its predecessors. It employs a pipeline organization feature to provide concurrent fetch, decode, and execute operations in order to increase the number of instructions executed per unit of time. All major FCC elements are provided with hardware error detection features to aid in the failure isolation and fault recovery process.

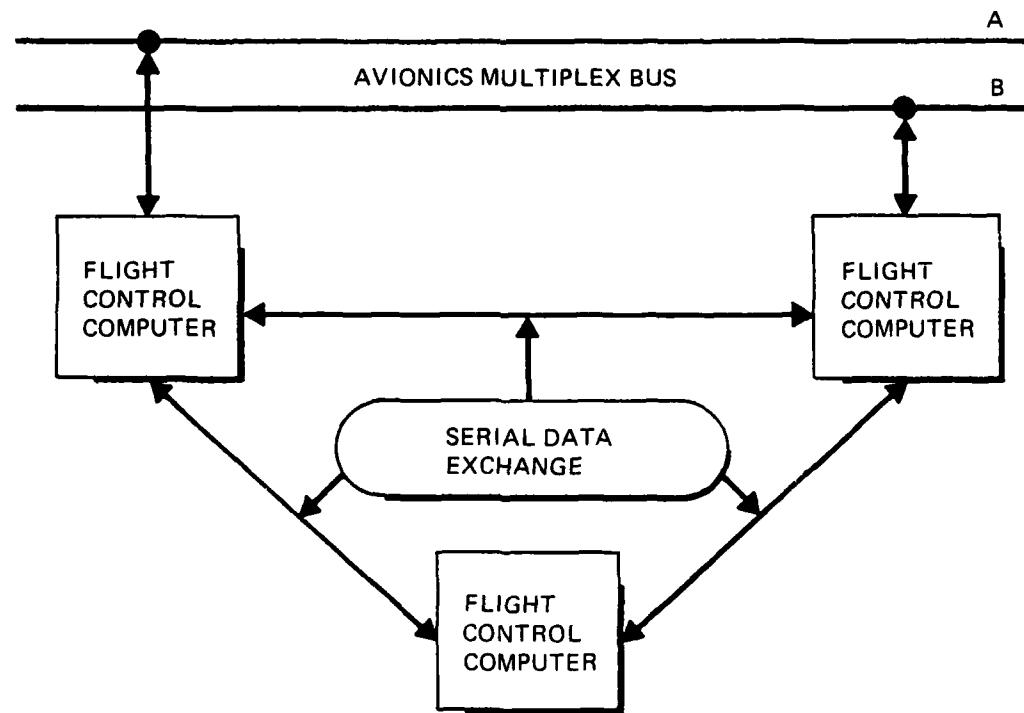


Figure 37. Interface of the FCC Complex with the Dualized Avionics Multiplex Bus

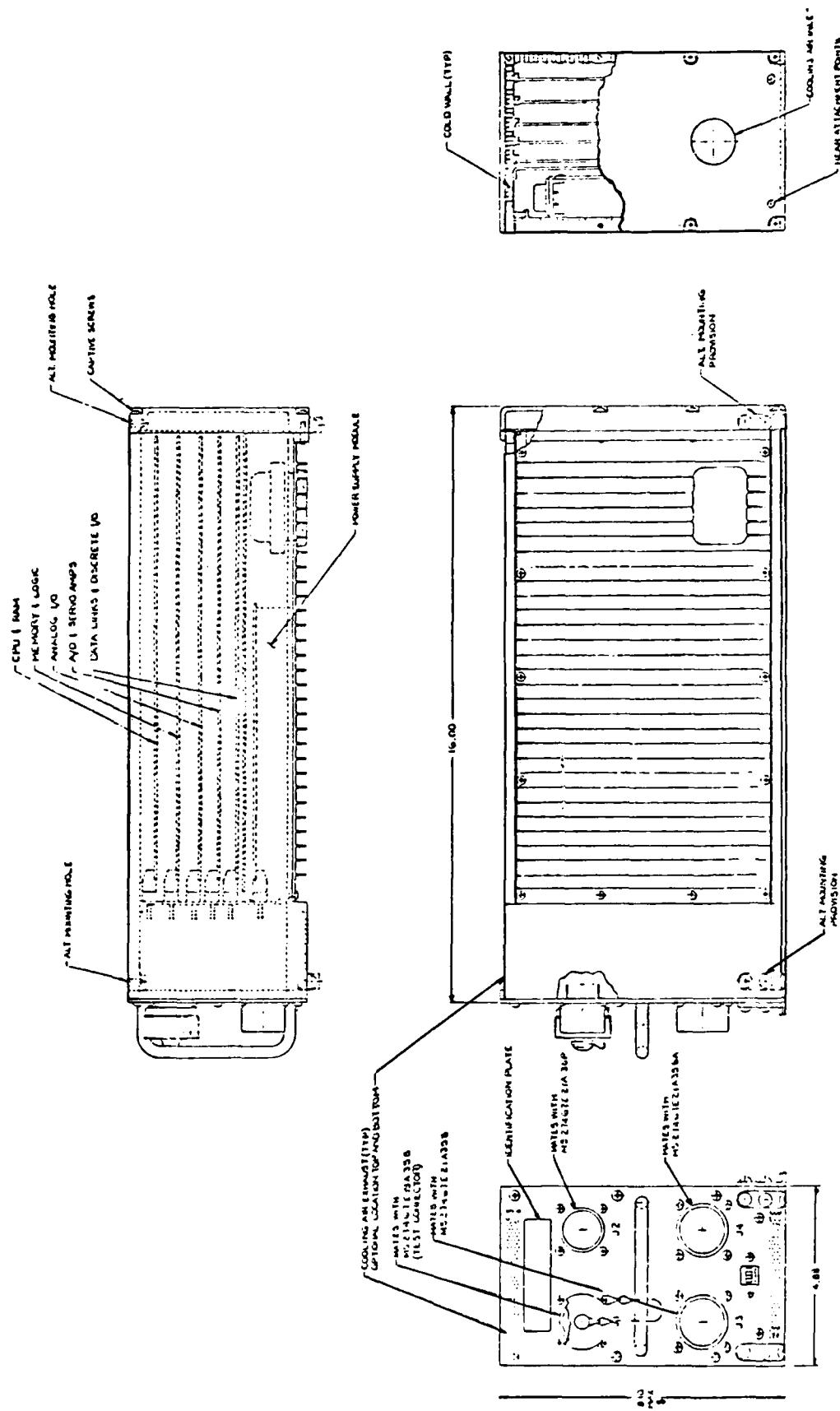


Figure 38. Layout/Outline Drawing of the Bendix BDX-930 FCC

The Central Processor Unit (CPU) architecture employs an asynchronous memory interface that allows the CPU (I/O) controller, avionics MUX bus, and inter FCC data link from loading down the memory access bus.

The processor can perform single precision, double precision, and saturated arithmetic operations. Saturated arithmetic capability minimizes fixed-point arithmetic overflow problems. All memory is equipped with parity, and program memory is accomplished in EPROMS to preclude unintentional alteration. The processor contains 22 registers which are usable by the programmer.

The FCC contains an Input/Output (I/O) controller that controls the I/O operation of analog I/O, discrete I/O, and inter FCC serial data transmission. Branch failure partitioning will permit a CPU failure to cause the I/O controller to obtain analog and discrete inputs and to transmit these data to the other FCC's so that input redundancy is maintained. The analog I/O function is implemented with 12-bit analog-to-digital (A/D) and digital-to-analog (D/A) converters.

The FCC contains a serial data transmitter and two receivers, which are used to perform the inter-FCC data exchange. The bus format is one way communication with a Manchester bi-phase level code and is accomplished at a 1-MHz bit rate. The transmitters and receivers contain hardware to verify the validity of all messages and to aid in the failure isolation process. The bus structure will support 128 messages with up to 7 data words each. Each receiver bus access is limited to 1 K of RAM to prevent address errors from propagating into the processor work area and destroying in-line data computation.

The avionics MUX bus interface employs a single transmitter receiver, which is interfaced directly with RAM, with access limited in hardware to 1K words. This limited access feature also prevents MUX interface failures from propagating into the FCC. The interface contains a tag timer that can be reset by the bus controller and read by the FCC to obtain a time correlated data set.

The FCC also contains considerable hardware logic, which is used in the fault isolation, identification, and self-testing process. This logic includes hardware that permits two good FCC's to "shut down" a third by mutual consent of the two good FCC's without dependence upon any action of the third "failed" FCC's processor or software (this is first failure logic). The first failure of an FCC causes the issuance of a discrete, which enables (unmasks) a self-test interrupt handling routine. When a second FCC failure is detected, the FCC that detects the failure issues a "start self-test" interrupt, which causes the FCC's to execute the self test routines in an attempt to isolate the problem. The first FCC to declare a self-test failure is "shut down", and the system continues to operate on the last good FCC (this is second failure logic). Logic is wired so the last remaining processor will not be disconnected even if it also "fails" self test. During the early part of the flight test program, the DFCS will include a simple triplex analog Independent Backup Unit (IBU) control system that will automatically be engaged if both of the remaining FCC's either pass or fail self test. This feature is authorized only if an IBU is installed, otherwise, the failure logic described above will apply.

Each FCC contains a branch of analog electronics for the IBU function and is used as a dissimilar backup to the digital computation function, and is used primarily to protect against generic software faults in the system. This feature was added because the purpose of the AFTI/F-16 program is to provide a test bed for new flight control concepts and their integration into a total weapons system. Since it is a test bed, it will undergo many software modifications. The IBU can be engaged manually with a switch on the SSC. The IBU is also automatically engaged whenever a second FCC failure can not be isolated by self test or when all three FCC's have been declared failed.

The FCC's possess a Built-In-Test (BIT) capability for preflight and maintenance testing. The preflight BIT is capable of isolating failures to a FCS line replaceable unit (LRU) to a 95% confidence level. No special Ground Service Equipment (GSE) is required and the need for pilot participation is minimized. Results of tests are shown on the Multi-Purpose Displays (MPD). BIT pre-flight testing functions cannot be engaged in flight. The BIT function tests all software, sensors, and actuators. After the satisfactory completion of preflight BIT the cockpit controllers and switches are tested as the pilot goes through the normal "sweeps" through these devices while on the ground.

Automatic maintenance BIT functions are identical to those of the preflight BIT. Test results are displayed on the MPD's which also provide a built in "multimeter" function. Fifteen thousand test points are automatically checked during the maintenance BIT procedure (5000 for each FCC). Manual test procedures can be configured on previous failure histories. The BIT maintenance testing function cannot be engaged in flight. Table 6 lists the salient features of the FCC's.

TABLE 6. SALIENT FEATURES OF THE BENDIX BDX-930 FCC

- PROCESSOR
 - 16-Bit Microprocessor
 - Pipeline Architecture
 - 22 Usable Registers
 - Single Precision, Double Precision and Saturate Arithmetic
 - 162 Microsec Benchmark Program
 - Expanded Instruction Set
- MEMORY
 - Asynchronous Memory Control
 - 7k RAM Scratchpad
 - 32k EPROM Program
 - 64 Words EROM - Failure Record
- ATTRACTIVE FEATURES
 - Analog IBU - Independent of Digital Processor
 - Self Contained DC Power Supplies
 - Two Real Time Clocks
 - 12-Bit A/D and D/A Converters

3. The Actuator Interface Unit

The Actuator Interface Unit (AIU) shown in Figure 39, houses those components of the FCS that are not triply redundant. These components include the dual redundant Leading-Edge Flap (LEF) interface, the fourth servo-amplifier for each ISA, and several simplex-to-triplex Junction Box type of functions.

The caution, and warning, IBU engage, and the stall warning light drivers are designed so that two discrete outputs are required for illumination. This feature makes it extremely remote that the lamps will be inadvertently illuminated by a failure.

The LEF circuitry contains a switching matrix that is controlled by the FCC's to determine which FCC is driving the LEF. The electronics also include summing amplifiers that close the rate and position loops around the LEF drive motors. The rate and position information is also transmitted to each FCC for monitoring purposes. Switches that deactivate either or both LEF drivemotors are also contained in the AIU.

A switching matrix driven by discrete outputs from the three FCC's is used to select which FCC drives the fourth servo amplifier for each ISA and to determine which back-up ISA valve coil is driven by this amplifier. The valve current wrap-around is supplied to all FCC's to permit monitoring.

The Junction Box functions of the AIU include the conversion of ISA pressure switch positions and ISA positions from simplex to triplex form. The "pseudo" position of each ISA is brought into the AIU where the signal is demodulated into a DC signal and then sent to each FCC for ISA monitoring purposes. Figure 40 shows the integration of the AIU within the DFCS.

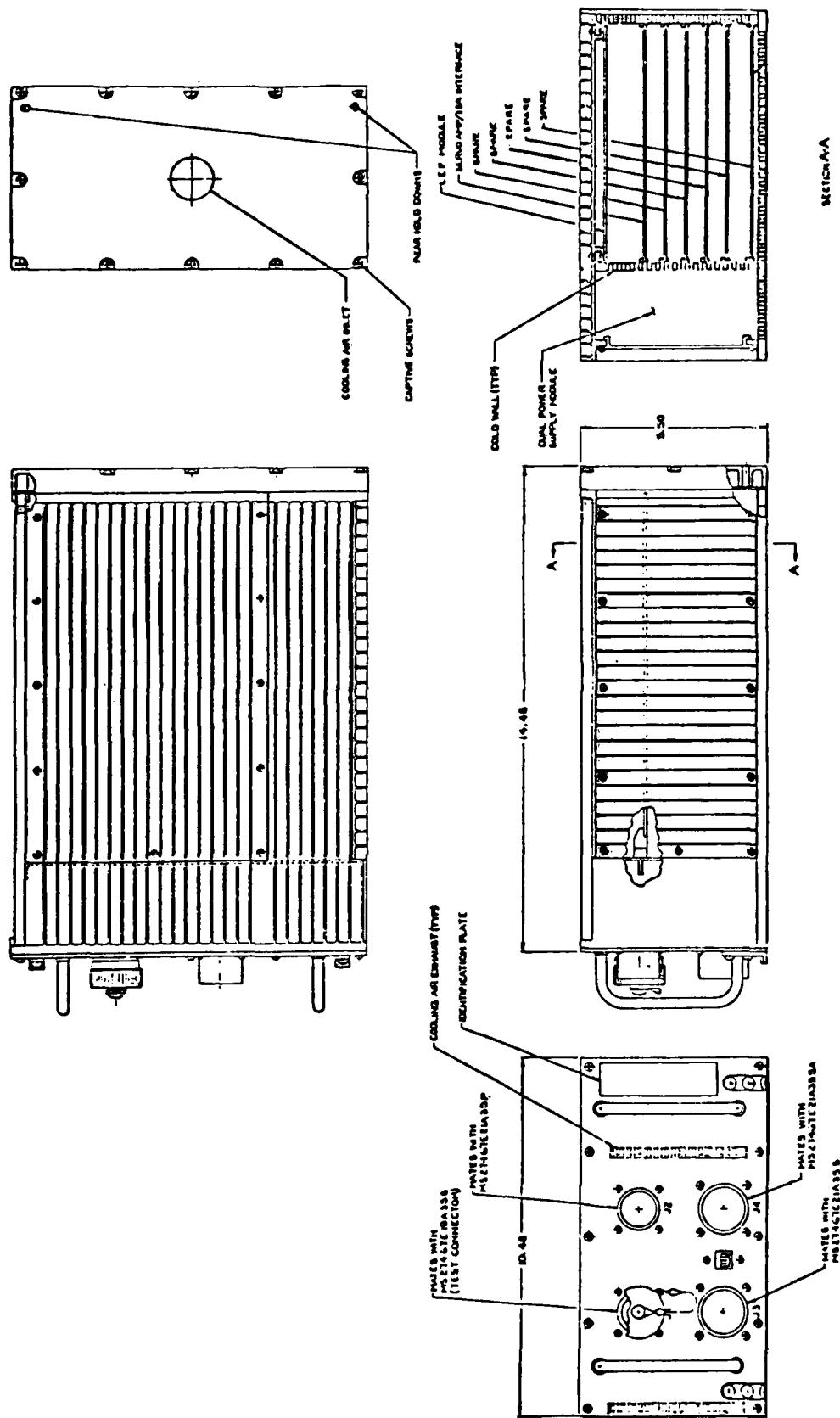


Figure 39. Layout/Outline Drawing of the Actuator Interface Unit (AIU)

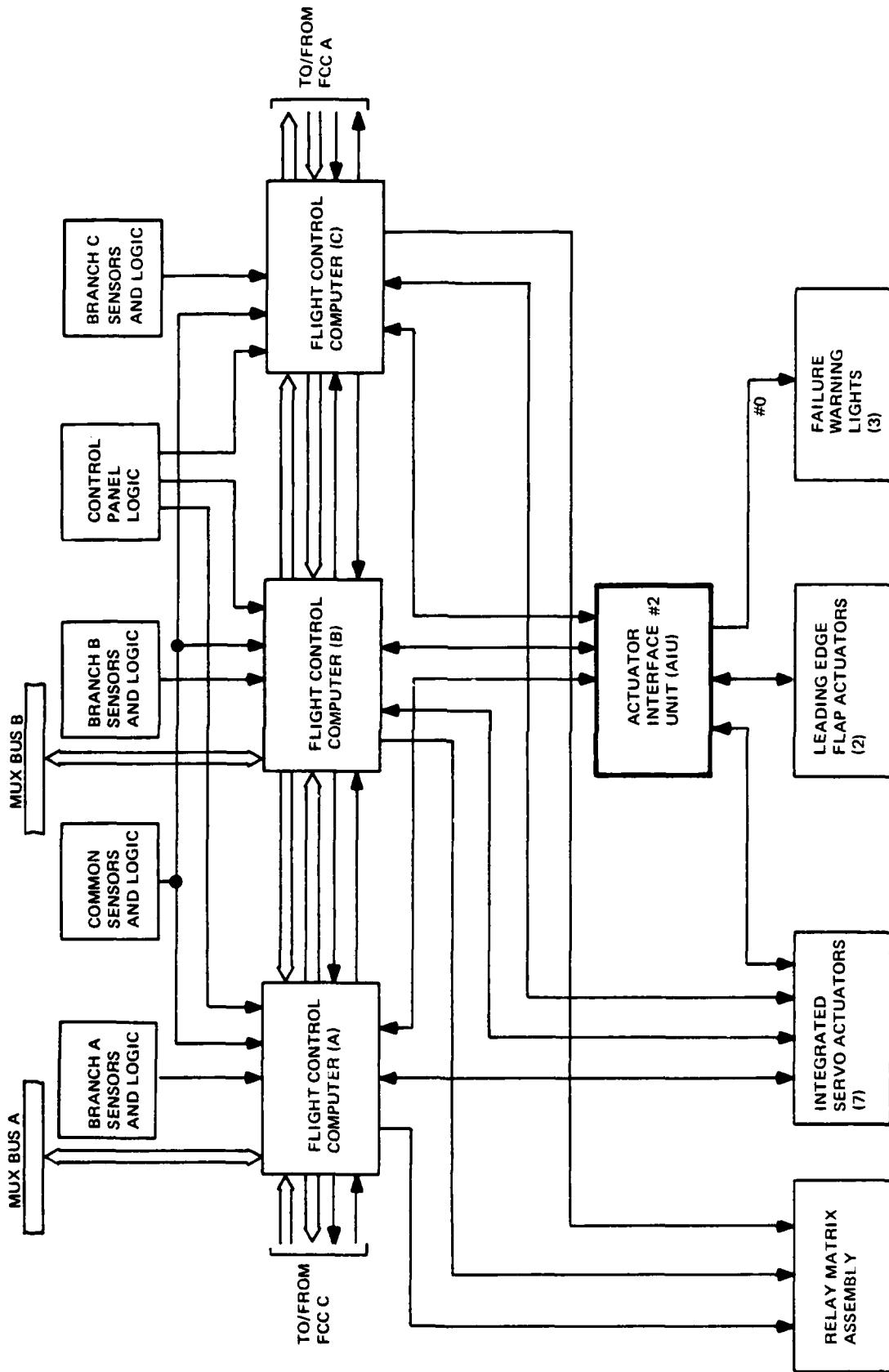


Figure 40. Integration of the Actuator Interface Unit (AIU) Within the DFCS

FLIGHT CONTROL ACTUATORS

This section describes the Integrated Servoactuators (ISA's) that drive the primary flight control surfaces and the Power Drive Unit (PDU) assembly that actuates the Leading Edge Flaps (LEF). The speedbrake actuators, part of the secondary FCS, do not interface with any other portion of the primary or secondary FCS and therefore will not be described.

1. Integrated Servoactuators (ISA)

Each of the seven ISA's (Figure 41) accept electrical commands from the FCC's in three electro-hydraulic servovalves (EHSV). These commands are converted into a power-ram position, which then positions the respective flight control surface. Actual ram position is fed back to the EHSV's mechanically to close the command loop. This mechanical feedback concept provides for improved surface position resolution and allows optimum ISA packaging.

A schematic diagram of the ISA is shown in Figure 42 to illustrate the following functional characteristics:

- A unique mechanical position and actuator rate feedback scheme combines the feedbacks into a single input to EHSV.
- Three EHSV's are provided for redundancy purposes. EHSV SV1 and SV2 normally share control of actuator position, while the third EHSV, SV3, is held in standby. Note that flows from SV1 and SV2 flow to the larger piston, and its chamber, on the main control valve. The flows from SV1 and SV2 are summed in the chamber.

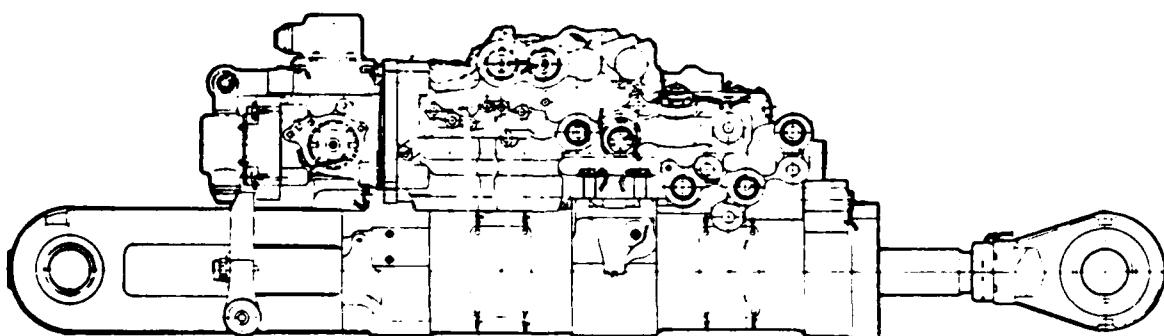


Figure 41. Typical ISA Used for Powering the Primary and Vertical Canard Control Surfaces

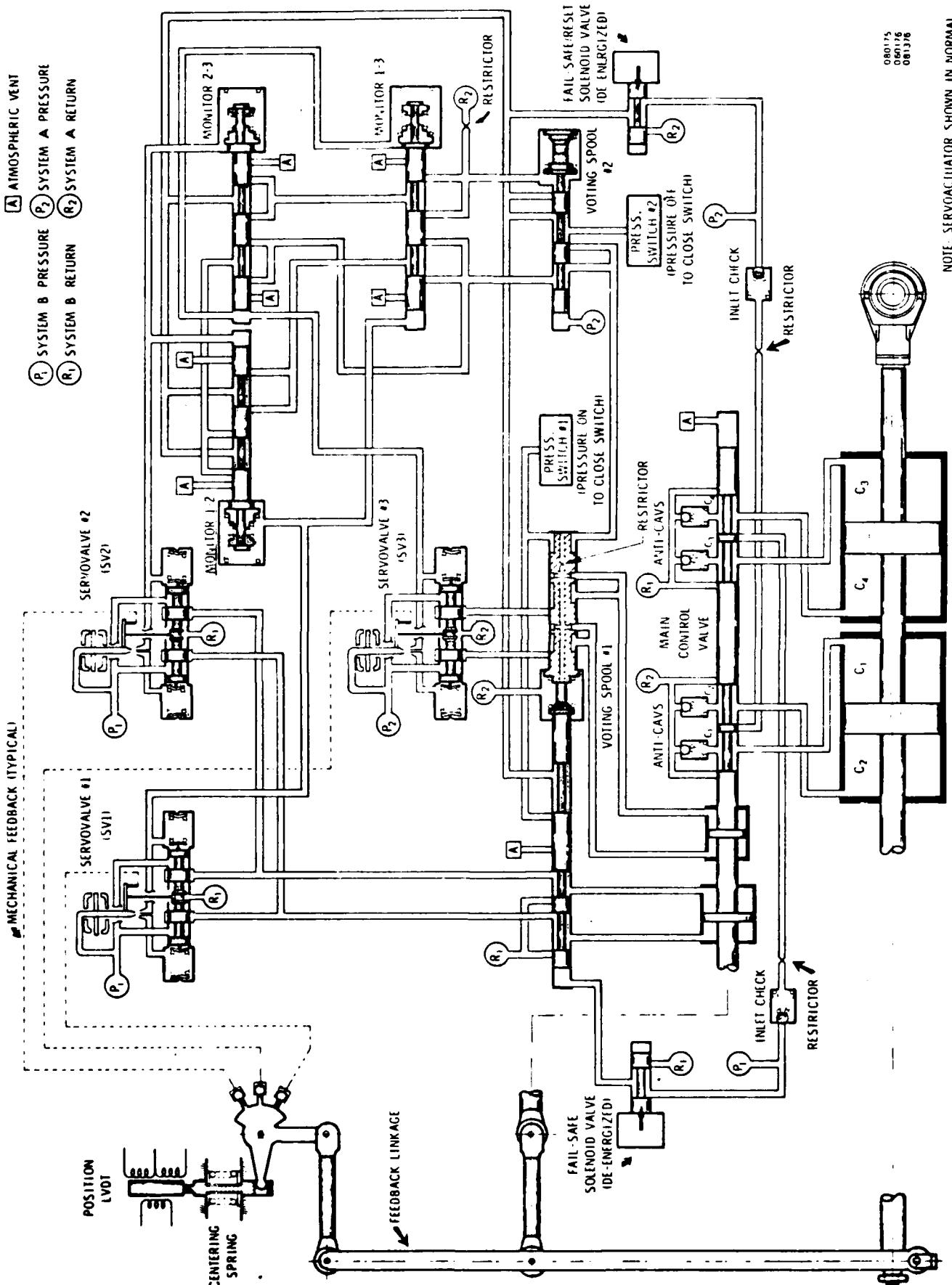


Figure 42. Schematic Diagram of the ISA

NOTE: SERVOACTUATOR SHOWN IN NORMAL OPERATING CONDITION

- EHSV failure detection is accomplished by comparing EHSV first-stage pressures.
- Self-contained hydromechanical failure detection and correction logic is incorporated for first failures of the EHSV's or for the hydraulic system. A first failure of SV1 or SV2 will cause transfer of control to the standby EHSV SV3. A first failure of SV3 will lock the ISA on SV1 and SV2 control.
- Hydraulic system failure correction is given precedence over all EHSV failures. EHSV's SV1 and SV2 operate on one hydraulic system, and SV3 operates on the other hydraulic system. One hydraulic system (System B) is connected to pressure P_1 and R_1 and the other (System A) is connected to pressures P_2 and R_2 .
- Fail-safe capability is incorporated to allow the ISA to mechanically center upon receipt of an electrical command to the fail-safe solenoids from an external electronic model and monitor unit.

In the mechanization of its hydraulic-mechanical redundancy management, the ISA utilizes self-contained failure logic to provide no less than one-failure-operative performance. With the aid of FCC monitoring, the ISA provides two-failure-operative performance if one of the two failures is electrical. By arming the outer loop monitor, the ISA provides fail-safe performance for multiple failures. The voting procedure is described below:

1. Hydraulic System Failure Detection. Hydraulic system voting logic is built into the ISA. This logic takes precedence over EHSV voting logic and will cause the ISA to select a good hydraulic system.
2. ISA EHSV Failure Detection. Each ISA uses three EHSV's. Normally, two of these EHSV's (SV1 and SV2) share control of actuator output position. If a failure occurs in either of these EHSV's the third EHSV (SV3) will assume control of actuator output position.
3. Flight Control Computer EHSV Failure Detection. Each EHSV has two windings, each of which has total EHSV command capability. These windings are driven by the FCC in an active standby manner so that a servoamplifier first failure will be detected and isolated and a spare servoamplifier in the AIU will be activated. This procedure will prevent the needless activation of the ISA self-contained voting logic and the loss of a level of redundancy in the EHSV control network. Figure 43 shows the interface of an ISA with the FCC complex and how a failed servoamplifier is bypassed and replaced with a spare servoamplifier in the AIU.
4. ISA Position Versus Computer Model Position. Each FCC models the ISA and compares the model output with the ISA position. If the two differ by a prescribed amount after a time delay, then an ISA fail discrete signal is outputted to the AIU. This modeling and monitoring is accomplished in software.
5. Mechanical Centering and Locking of a Failed ISA. Additional failure protection capability is designed into the ISA so that it can be mechanically centered by bypass commands from the FCC's. These commands cause all EHSV's to be bypassed and allow a self-contained spring to mechanically command the ISA to a predetermined neutral surface position. Centering can only occur if (1) the pilot has previously armed the outer loop monitor, and (2) the outer loop monitor has determined that the surface position is in error. This action prevents spurious failure. The ISA should be centered after failures of either SV1 or SV2, and failure of SV3. A combined failure of SV1 and SV2 is tolerable provided that SV3 and Hydraulic System A are operational.

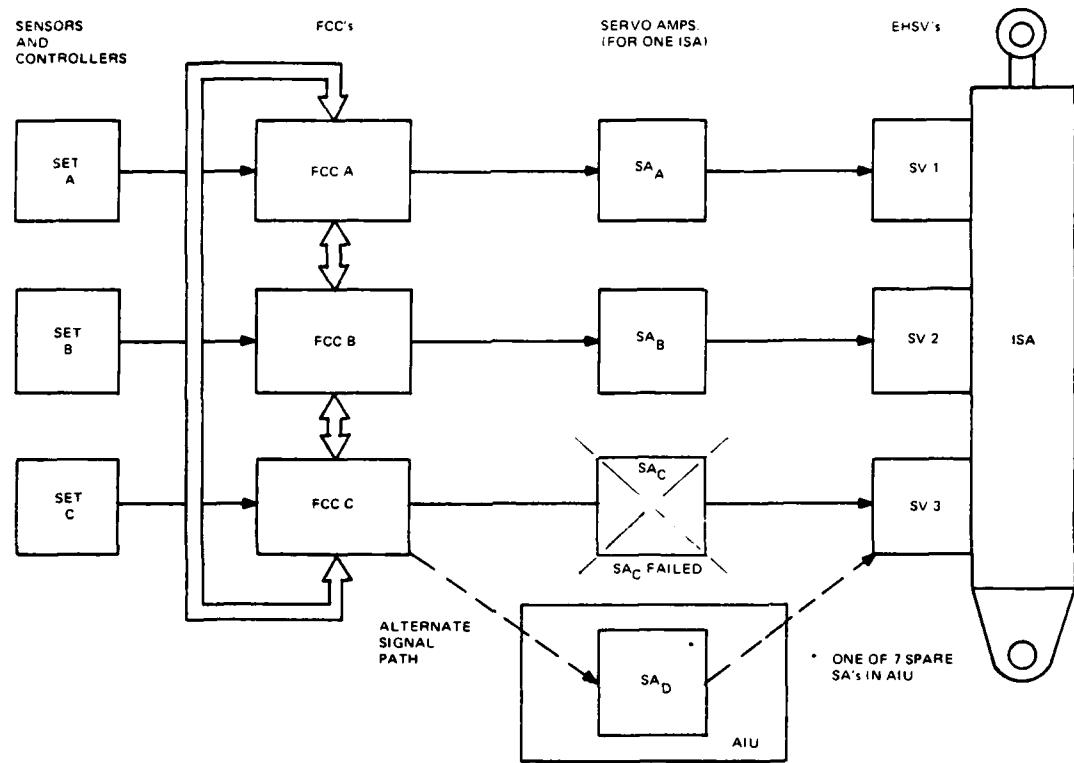


Figure 43. Interface of an ISA with FCC Complex

2. Power Drive Unit (PDU) Assembly

The Power Drive Unit (PDU) assembly shown in Figure 44 is comprised of the basic hydromechanical drive unit plus the following components:

- A hydraulic flow-control valve and manifold to drive both hydraulic motors.
- Flow limiters to limit speeds for low loads on the LEF.
- A redundant electromechanical actuator to provide a positioning command to the hydromechanical drive unit.
- Position summing linkage.
- Operational controls.

Two in-line hydraulic motors (one driven by each hydraulic system) power the LEF. These motors are located on opposite sides of the PDU to provide maximum separation of the two independent hydraulic systems. Two hydraulic motors, ganged together by a common shaft and pinion, drive a bull gear. The bull gear, in turn, drives two output shafts; one on the left side, another on the right side. The shaft on the left side contains gearing that drives a feedback arm through a 60-degree arc of the electromechanical actuator's rotating crank.

The shaft on the right side contains an overtravel mechanism that allows one degree of additional surface motion at each end of the operating stroke of the panel. After contacting the stop, the system will come to a halt within two output shaft revolutions of the PDU. This safety feature provides internal stops that prevent the LEF from being driven into hard structure in the event of a malfunction.

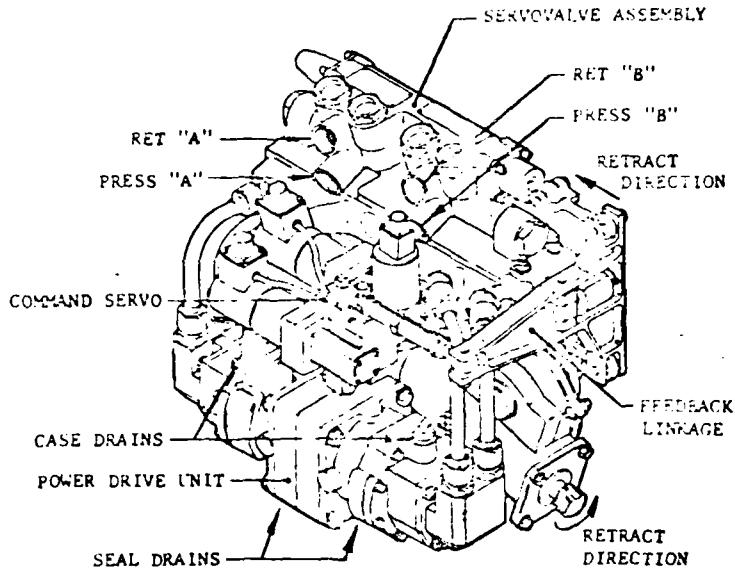


Figure 44. Power Drive Unit (PDU) Assembly

in either the control or drive portion of the PDU. The positive stops consist of a threaded nut, which moves inboard and outboard as the shaft rotates, and brake discs held with bellville-type washers. Should the PDU tend to overtravel, the traveling nut engages the spring washers and compresses them, producing a clamping action that stops shaft rotation and stalls the motor.

REDUNDANCY MANAGEMENT OF THE DFCS

The reliability and fail-operational requirements for the AFTI/F-16 DFCS are listed as follows:

- Fail Operational Requirements
 - First Failure — Undergraded full operational performance
 - Second Failure — Safe flight capability
 - Fully operational (State 1) performance 95% probable
- Reliability Requirements
 - Loss of Control — 1 in 10^7 flight hours
 - Mission Abort — 1 in 10^5

To fulfill these requirements the DFCS must possess redundancy management techniques capable of providing optimum failure survivability via detection and isolation of failed components to the LRU level and reconfiguring the remaining unfailed components to provide the maximum level of aircraft safety and the highest probability of mission completion.

To meet the fail-operational and reliability requirements the DFCS has three voting/monitoring planes as shown in Figure 45. Two of these planes are in software and are at the sensor/controller interface and the output flight control surface command interface. The hardware voting plane is located internally to the ISA's and can be used to isolate failures associated with the FCC output circuitry and ISA EHSV coils as well as internal ISA failures.

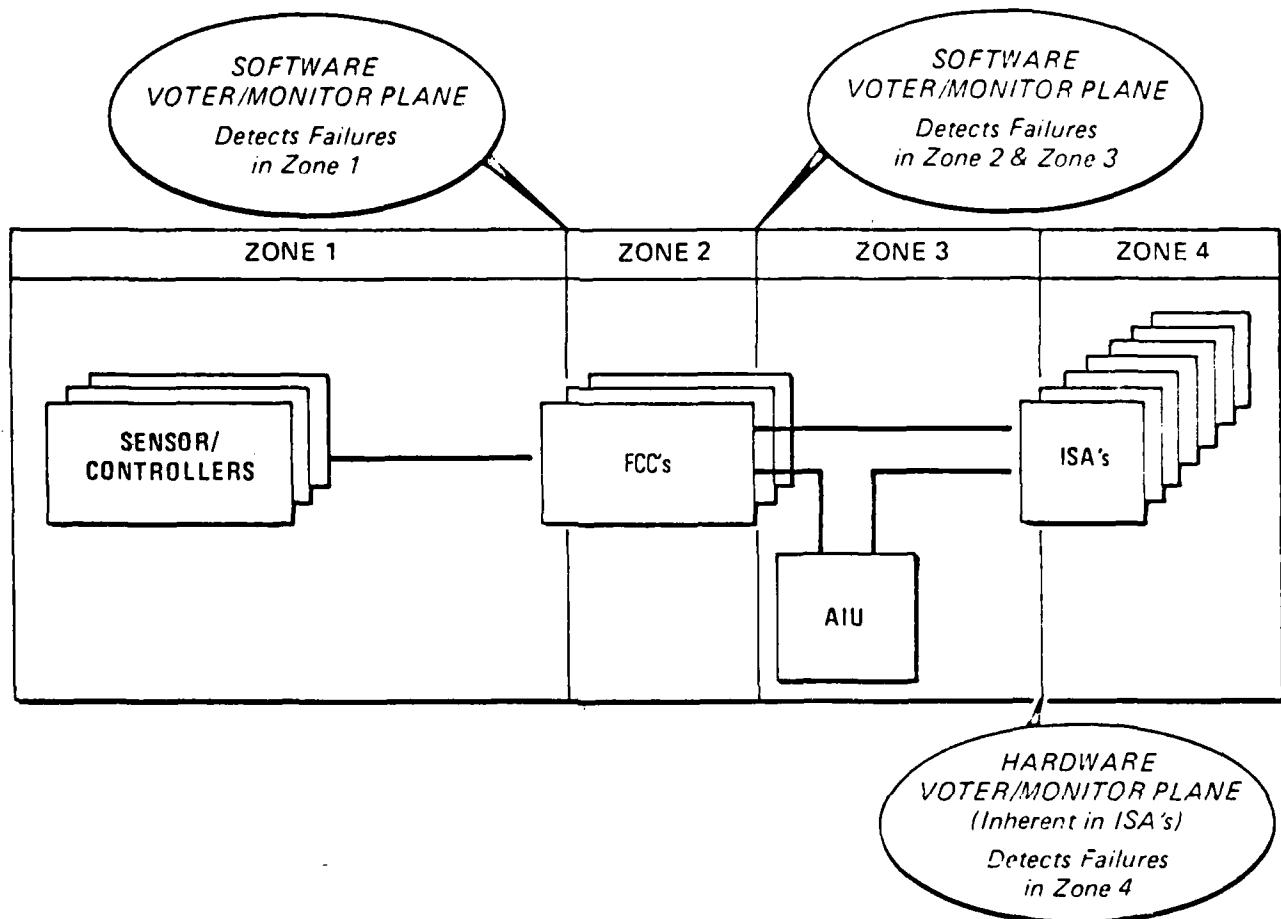


Figure 45. Redundancy Data Management of Hardware and Software Functions of the AFTI/F-16 DFCS

The general guidelines in evolving redundancy management techniques are listed as follows:

- No complete dependence for a FCC to recognize that it has failed after performing a self-test routine.
- Minimize "nuisance" alarms by
 - Placing maximum reliance on comparison monitoring
 - Provide safeguards against transitory failures
 - Use self-test methods only where required
- Implement a safe, dependable failure detection and identification scheme.
- Provide a failure survival capability as good as the present F-16 analog FBW system.

The redundancy management techniques are significantly different at each of the three voting/monitoring planes and will therefore be discussed separately.

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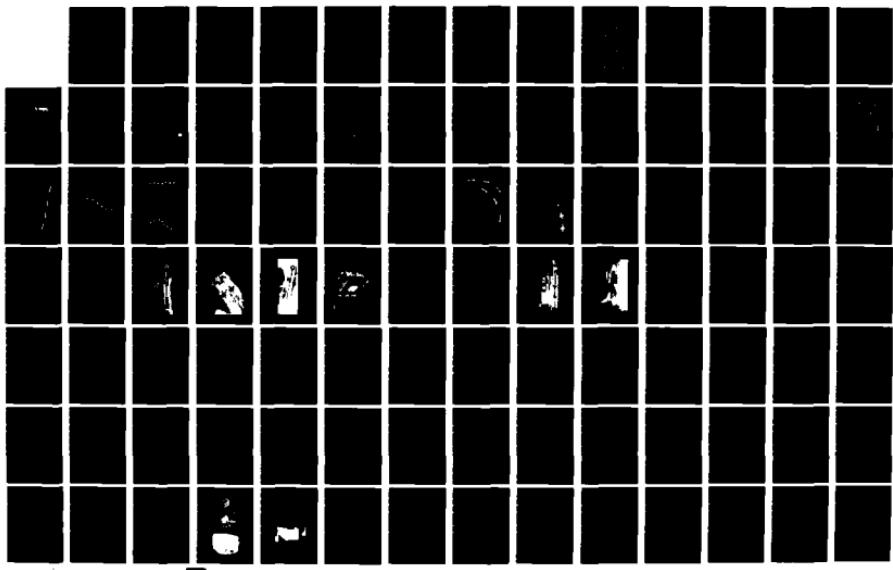
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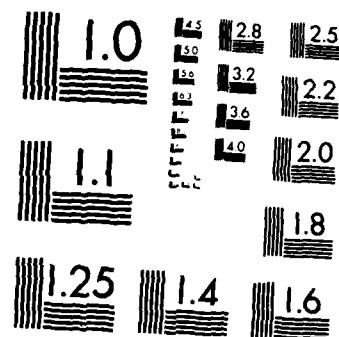
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1. Input Voting/Monitoring Plane

The primary purpose of the input voting/monitoring plane is to detect and isolate failures associated with the sensors, controllers, and input circuitry. Redundant analog and discrete inputs are supplied to each FCC with the required cross-strap for redundant data selection implemented by a serial data exchange between FCC's.

During normal (no failure) operation input data are acquired from pilot inputs, motion sensors, and ISA position signals, at a rate of 256 samples/second. The remaining FCS cyclic input data are acquired at a rate of 64 samples/second. These data are sequentially received by the Input/Output Controller (IOC) of an FCC, deposited in an area of scratchpad Random Access Memory (RAM) dedicated to the IOC, and simultaneously transmitted to the other two FCC's. The IOC's are micro-sequenced controllers designed to operate independently of the FCC Central Processing Unit (CPU).

Whenever an FCC is declared failed, the redundancy management hardware issues discretes, indicating the failure, to the IOC of the failed FCC as well as to other branches. The discretes cause the IOC of the failed FCC to execute its backup (free run) file and continue to supply input data to the other FCC independently of any command from the failed FCC.

Each FCC performs input voting and monitoring on the redundant input data deposited in its scratchpad memory by its IOC as well as by the IOC's of the other two FCC's. Two selector/monitor algorithms are used to perform the input voter/monitor function. One for discrete input and one for analog input data.

Discrete inputs are monitored before voting and one monitoring cycle is permitted (persistence) before a disagreement is declared a failure. Discrete inputs are monitored and voted in digital packed word form and each digital word may contain as many as sixteen discrete inputs. Most discretes are monitored at a rate of 64 samples/second but discretes affected by mechanical contact bounce are monitored at a rate of 13 samples/second.

The selector algorithm for discrete inputs performs a majority vote if the three discretes agree or if one of the three disagrees (a possible first failure condition). If a first failure is declared and the two remaining discretes disagree, (a possible second failure condition), the selector algorithm selects the last value used prior to the disagreement. If the disagreement cannot be resolved, a failure is declared and the most safe state for the discrete is assumed and selected.

All analog input signals are monitored and selected. AOA and AOS signals are normalized and compensated for errors prior to utilization.

A good channel average algorithm operates on all analog inputs, (except AOA and pneumatic Beta which use a modified median select algorithm) and selects the average value when signals from either two or three good channels are present. If any signal does not track the other two sensor signals within a band of tolerances, that sensor signal is not included in the average. If a sensor signal does track for a number of consecutive frames, the failure management function is called to remove it from the input sensor set. After a first analog input failure, the failure management function is called to resolve any disagreement between the last two good channels (a possible second failure condition). If the second failure can be isolated, the remaining good channel signal is selected. If the second failure cannot be isolated, a reconfigured set of control laws is provided.

Reconfigured control laws are available for all analog inputs except controller commands. When reconfiguration is required for the pilot sidestick and rudder pedal inputs, the signal from the appropriate fourth force transducer on these input devices is used.

The averaging process associated with the good channel average algorithm eliminates sensor state differences. All inputs which either change rapidly or are significant contributors to gain and phase margins are acquired at a rate of 256 samples/second to minimize data latency.

2. Output Voting/Monitoring Plane

The output voting/monitoring is accomplished by the ISA monitor and the LEF monitor. These monitors employ a combination of FCC software and hardware to manage the output and system interfaces.

The ISA monitor is designed on the assumption that all critical FCC and output electronic failures can be detected by compiling the coil currents (obtained via analog wraparounds) which drive the three EHSV's in each of the seven ISA's. Because of the number of ISA's and the complexity of the monitoring scheme the ISA monitoring function is broken into a subframe ISA monitor which executes every 8 milliseconds and a Frame Cyclic ISA monitor which executes once per frame (16 milliseconds).

The subframe ISA monitor is given the task of comparing the coil currents of the primary ISA coils. With no failures in the system FCC A drives the primary coil of SV1, FCC B drives the primary coil of SV2, and FCC C drives the primary coil of SV3 with the average displacement of SV1 and SV2 actually driving the surface via the primary hydraulic system (see Figure 43). An erroneous command from an FCC or a failure of the output electronics (D/A, sample and hold, servoamplifier, or primary coil) will cause a difference to exist between the three primary coil currents. Each FCC (if not the cause for the difference) is capable of isolating the failure to a particular branch. If an FCC determines that its coil current is erroneous the output electronics are disconnected and a backup coil current is supplied using the fourth servoamplifier driven from a different branch. If this procedure is accomplished successfully the other two FCC's allow the partially failed branch to remain on line. If the branch with the failure does not disconnect the primary coil, the other two branches will totally disconnect the failed branch. The task of isolating this failure to either a computed output failure or an output electronics failure is left to the ISA frame cyclic monitor for first failures.

The Subframe ISA monitor becomes quite complicated after a first failure has been detected and isolated. The resulting system actions are dependent on the cause of the first failure (output electronics or erroneous FCC output), the cause of the second failure, whether or not the first or second failure affected one or all coil currents, and which branches sustained the failures.

The Frame Cyclic ISA monitor is allocated the tasks of monitoring the backup (or 4th) ISA servoamplifiers, handling the automatic reengagement of primary servoamplifiers which have been taken off line due to a transient condition, monitoring the ISA centering and pressure system discretes, monitoring the computed ISA commands after a first output electronics failure, and controlling the activation and computations associated with the ISA model which can be engaged after an ISA mechanical or hydraulic system failure.

The LEF monitor performs cross-channel monitoring on the LEF commands, position transducer signals, and LEF motor rates. The LEF monitor also contains a digital model of the LEF actuation system. The digital model is modified to reflect full or half speed drive as a function of the LEF failure state.

a. Failure Detection, Isolation, and Reconfiguration After Single and Dual FCC Failures

As stated previously each FCC receives the ISA commands of the other FCC's via the intercomputer data link. Each FCC, by comparison monitoring of ISA commands decides if another FCC was failed in any (or all) of its ISA commands. For the single failed FCC situation we will assume that FCC A experienced a failure. FCC A is expected to detect its own failures by comparing actual ISA position with that of the model ISA's, and shut itself down if these values disagree. FCC B and FCC C monitor FCC A's ability to monitor itself.

If FCC's B and C detect a failure of one (or more) of FCC A's ISA commands, they wait to see if FCC A identifies the failure and takes appropriate action.

If FCC's B and C receive intelligence via the inter FCC data link that FCC A has taken the proper action, then FCC's B and C take no action.

If however, FCC's B and C do not receive intelligence that FCC A has taken the appropriate action (in connection with an FCC A failure they have detected) then they assume that FCC A has failed and should be switched out or isolated from the system. In this case, FCC's B and C send a discrete to FCC A to indicate a failure. If both FCC B and C send such a discrete, then discrete failure logic in FCC A switches FCC A out of the system. The failure discrete which FCC's B and C send to FCC A is not associated with a particular ISA, but rather implies a failure of FCC A as a whole.

The sequence of events which take place after a second FCC failure (FCC B or FCC C) will now be discussed.

If both FCC's (B and C) detect a failure in a particular ISA command, then both FCC's revert to previous value of the appropriate ISA command. Both FCC's will initiate self-test procedures. When the "failed FCC" is identified it is switched out of control of the affected ISA.

If only one of the two remaining FCC's detects a failure, then both FCC's revert to previous values for all ISA commands. Both FCC's initiate the self-testing procedures. When "failed FCC" is identified, it is switched out of control of all ISA's.

If both of the last two FCC's fail self-test, then the first to respond controls the FCC switching. The self-test sequence is arranged in priority order.

If neither of the last two FCC's fail self-test, but one FCC does not complete it (no pass or fail), then it is designated the failed FCC.

If neither of the last two FCC's fail self-test, and no other information is available for the decision, then one is arbitrarily designated the failed FCC based on a predetermined hierarchy.

If a second FCC failure is not isolated then the IBU is engaged. The IBU is also automatically engaged whenever all three FCC's indicate that they have failed.

3. Output ISA Voting/Monitoring Plane.

The primary purpose of the ISA voting/monitoring plane is to detect and isolate mechanical and hydraulic failures internal to the ISA. Since the ISA's are driven by a dual-redundant hydraulic system, they are designed to provide a fail-operate capability. In order to provide this capability, a first failure upstream of the ISA must be isolated before the ISA voter/monitor will detect this

same failure. The ISA monitor will declare a failure in approximately 20 to 40 milliseconds; therefore, the upstream monitors must remove a failure before this time limit. Because of this time limit, the upstream voter monitors will remove the failure on the first detection and will initiate the proper corrective action but not declare a failure for several cycles. This latency is desired because it reduces the tendency of the DFCS to declare hard failures during transient or nuisance failure conditions.

FLIGHT CONTROL SYSTEM SELF-TEST

Flight Control System (FCS) self-test capability can be viewed as consisting of four tiers of tests of increasing complexity and fault isolation coverage. These tests include (in order of increasing complexity) the following:

- Inflight Integrity Management (IFIM)
- Inflight Self-Test
- Preflight BIT
- Maintenance BIT

1. Inflight Integrity Management (IFIM)

IFIM is active whenever power is supplied to the FCS and is used to protect against only those failures which, if they occur in flight, would have a direct effect on FCS performance and/or cause a reduction in the redundancy level of the system.

The primary IFIM mechanisms are in the three voting/monitoring planes previously described. When failures are detected at these planes by the cross-channel monitors and persist, the pilot is informed of the failure by means of caution or warning lights and by a message on the Multi-Purpose Display (MPD). By keying up the FCS fault-report page on the display, the pilot will be informed of the exact failure which has occurred and can attempt a reset. A reset command will attempt to clear the failure which is on the MPD and has the effect of restoring the redundancy by one level. Thus a reset of a second-like failure only resets the second failure and cannot foil the voting algorithm. If a reset is successful, the appropriate message will disappear from the MPD and the lights will be extinguished.

IFIM also includes several hardware monitors which are also continuously active. These monitors perform the following functions:

- Monitoring of electrical power which will produce a high priority vectored interrupt in case of power failure.
- Watchdog timer function which disengages FCC outputs in case of major FCC failures.
- Monitoring of the word count and fresh data flags on the Avionics and inter-FCC MUX bus receivers.
- Monitoring of fresh data flags on A/D converter outputs.

The purpose of these hardware monitors is to declare a single hardware failure instead of a multitude of input or system failures as would be the case if only the selector/monitors were employed. The information from these hardware monitors in conjunction with the failure information supplied by the selector/monitors is used by a software routine called the "failure manager" to determine what failure message is supplied to the MPD's.

2. Inflight Self-Test

Inflight self-tests include hardware tests which are always active, software cyclic tests, and tests which are event-driven (usually a failure). A failure detected by these tests is always recorded by the FCC and transmitted via the MUX bus to the instrumentation system but does not necessarily result in a failure indication to the pilot.

Hardware self-tests, which are always active, include memory parity on all FCC memory and parity checks on all MUX bus receivers. A memory parity failure causes a processor interrupt and executes a software routine which records the failed location for later retrieval. Parity errors on transmissions are used to alert the FCC not to depend on the data received during that cycle.

CPU self-test, RAM, and ROM tests are event-driven and are executed in the air only after a second-like failure is detected at the output monitor plane or on the ground by a higher level test.

3. Preflight BIT

Preflight BIT includes all IFIM and inflight self-tests plus additional FCC tests and testing of LRU's external to the FCC. Preflight BIT is initiated, controlled, and monitored via the MPD's. The tests are performed by the FCC resident Operational Flight Plan (OFP) and can only be initiated on the ground (interlocked by appropriate discrete inputs).

When power is applied to the FCS, all FCC's are in the IBU mode and will start executing flight control laws associated with the NORMAL multimode when brought on line, with IFIM and inflight self-tests operational, and with the corresponding NORMAL mode display. The pilot will then check his failure lights and fault-report display and then initiate preflight BIT.

The preflight BIT display lists all manual tests which are still pending (not yet performed) and failure codes for all tests which were not passed.

4. Maintenance BIT

Maintenance BIT adds non-structured tests to those performed by preflight BIT in order to check out the FCS after maintenance or modification or as an aid in failure isolation to an LRU. All maintenance BIT tests are designed to use an assumed healthy FCC to find faults in exterior interfaces or equipment.

Maintenance BIT essentially turns the MPD's into remote computer terminals which allow the operator to look at the contents of any memory location in all three FCC's simultaneously and then use an MPD as a multimeter. For example, the operator could key in the memory location corresponding to left AOA, have a technician position the transducer to a predetermined position, and then determine if all electrical inputs were received by each FCC properly with the correct sign and within magnitude tolerances. By proper test selection, the operator can check out any system modification or isolate failures to an LRU.

INDEPENDENT BACK-UP UNIT (IBU)

To provide an independent backup capability, a very simple analog Independent Backup Unit (IBU) was developed. The IBU provides stability augmentation and control in pitch and direct flight control surface control in roll and yaw. A block diagram of the IBU is shown in Figure 46.

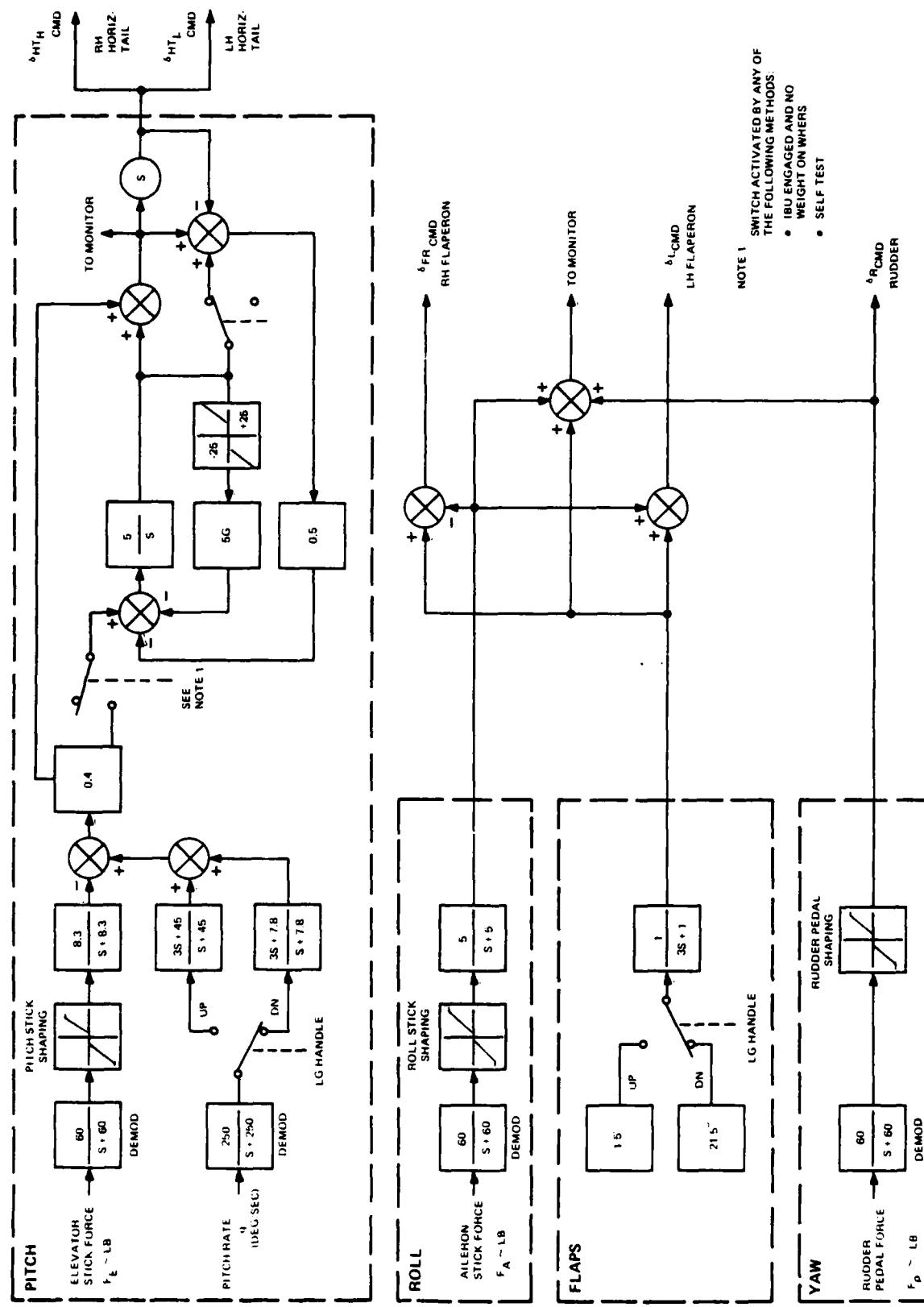


Figure 46. Block Diagram of the DFCS Independent Backup Unit (IBU)

The design philosophy for the IBU is to provide a simple controller and ensure safe flight. Performance requirements of the IBU are to meet Level 3 requirements of MIL-F-8785 (ASG) for cruise and descent and Level 2 for landing. Control in the longitudinal axis is provided by a pitch-rate command system. The pitch command from the stick is shaped and then lagged by a prefilter. The pitch rate error signal is then gained and fed through a proportional-plus-integral network. Different lead/lag compensators for the gear-up and gear-down configurations augment stability margin and improve handling qualities over the entire flight envelope. The forward loop integrator is not driven when the IBU is not engaged. The integrator is driven when the IBU is engaged and there is no weight on wheels. In order to minimize engagement transients, the integrator is also driven when the digital system is operating, but a self-test is being carried out. If the IBU is engaged, the integrator is discharged when there is weight on the main landing gear. Direct aileron control is used in the lateral axis. The command signal from the rudder pedal is simply passed through a shaping gradient function. Fixed biases are provided to drive the trailing-edge flaps. With the landing gear up, the flaps are driven to 0 deg. The leading-edge flaps are not driven by the IBU, but are simply locked once the IBU is engaged. Analysis has shown that the IBU satisfies Level 3 flying qualities for cruise and descent and Level 2 for landing. Stability margins in the pitch axis are adequate over the flight envelope. Pilot-in-the-loop simulation has shown that the IBU provides capability for a safe return to base and landing for emergency situations.

DFCS-ELECTRICAL SYSTEM INTERFACE

The FCS will receive electrical power from two essential 28 VDC buses, backed up by four batteries, a 5kVA emergency generator, and a 500 VA permanent magnet generator. The electrical system will provide uninterruptable, regulated power regardless of transient voltages or fault conditions elsewhere in the airplane system. This input power will be provided to each branch of the FCS where it will be finally converted and regulated to produce the 800 Hz AC and the DC voltages to power the DFCS. Figure 47 illustrates the AFTI/F-16 DFCS redundancy concept. Column 3 of this figure illustrates how the primary 40 kVA generator is backed up.

A schematic diagram of the AFTI/F-16 Electrical Power System is shown in Figure 48.

Note that each of the two essential 28 VDC busses receives its power from a dedicated 115/200 VAC-to-28-VDC converter. Each converter may receive input power from either the 40-kVA primary generator of the Emergency Power Unit (EPU) 5-kVA generator. The EPU is driven by engine high-pressure bleed air in the event of improper generator voltage or frequency. Should both the 40 kVA primary generator and the 5 kVA emergency generator fail to supply power to each one of the 28 VDC converters, then the Permanent Magnet Generator (PMG) portion of the EPU's emergency generator supplies 18-volt, 3-phase, 1,200 Hz, power to a third converter dedicated wholly for FCS power supply usage. Furthermore, this converter supplies power to the EPU's electronic controller so that continued operation of the EPU is assured (a "boat-strap" type of arrangement).

Multiple power sources preclude loss of power to the battery buses because of a malfunction in the input power system. The 24-volt batteries (18 cell, Ni-Cad type) prevents power interruptions during power source switching operations.

The electrical power supply system does not provide any "turn-off" capability to the pilot. However, provisions are included for shutting off electrical power to the FCS while the airplane is on the ground. Ground "shut-off" provisions are desirable from the standpoint of equipment life and reliability so that the FCS need not be energized during routine maintenance operations.

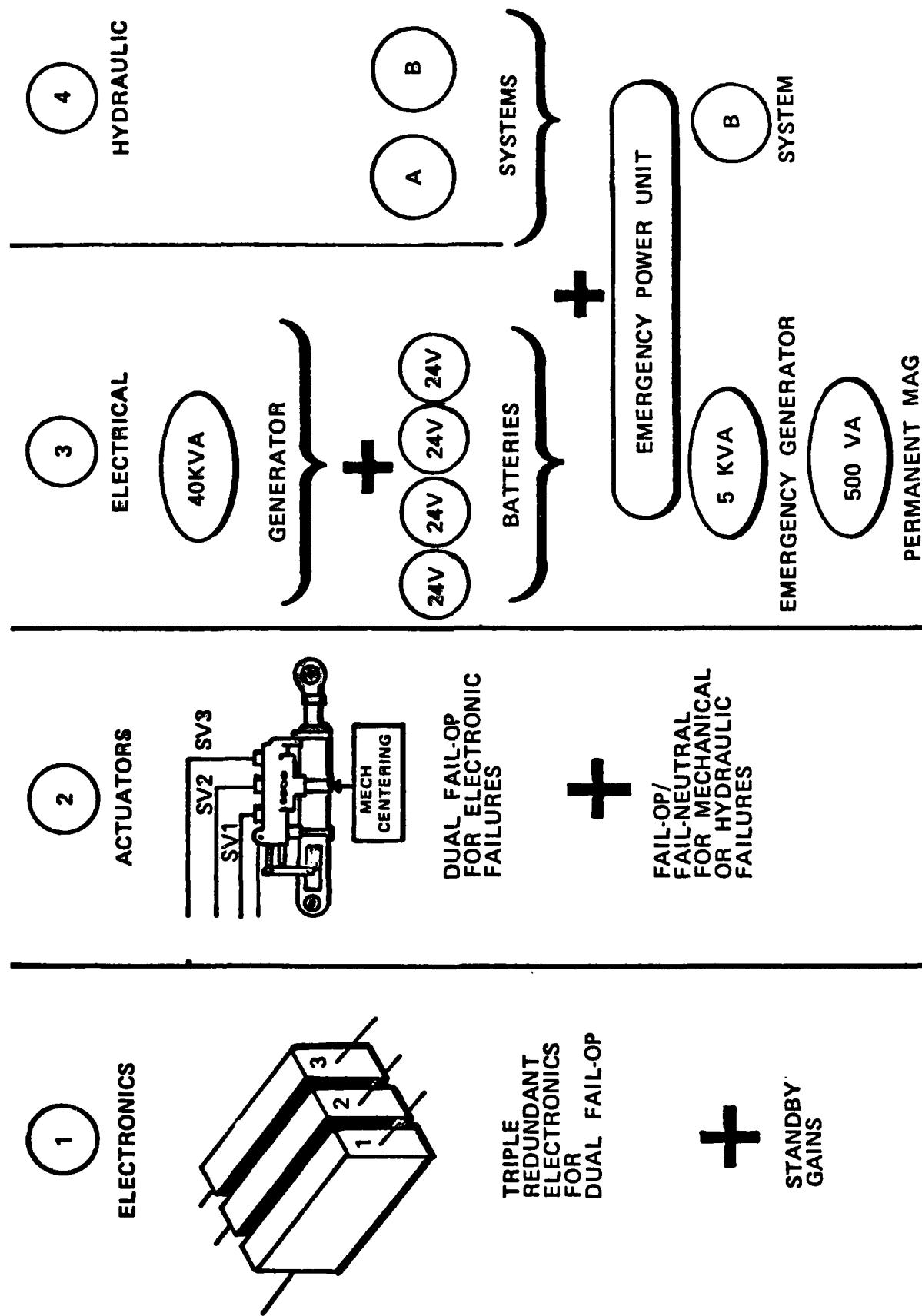


Figure 47. AFTI/F-16 Redundancy Concept

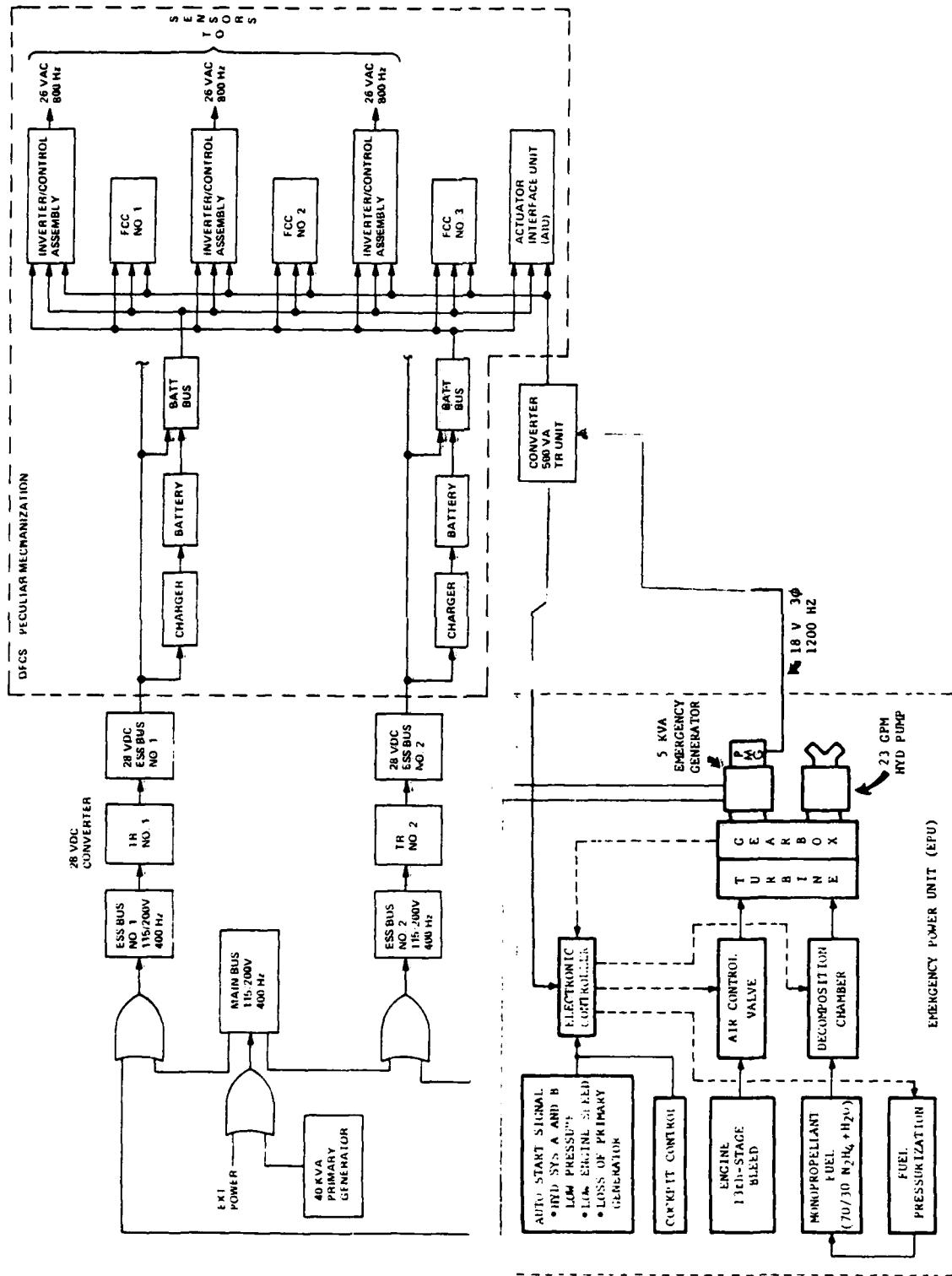


Figure 48. AFTI/F-16 DFCS Electrical Power System

DFCS-HYDRAULIC SYSTEM INTERFACE

Flight control surface actuation is dependent entirely upon the successful operation of the hydraulic system. The AFTI/F-16 hydraulic system consists of two separate and independent systems that supply hydraulic power for operation of the primary flight control surfaces, the vertical canards, LEF, speed brake, landing gear, nose wheel steering, wheel brakes, fuel flow proportioner, and other utility functions.

A block diagram of the hydraulic system is shown in Figure 49. The major elements of the hydraulic system consist of the following elements:

- Two main hydraulic pumps
- An emergency hydraulic pump
- Hydraulic reservoirs
- Seven flight control ISA's.

DFCS-AVIONICS SYSTEM INTERFACES

The DFCS and certain elements of the avionics system (see Figure 50) will be operated in aggregate to provide the integrated-DFCS-demonstrator functional capability. As shown in Figure 50 the DFCS is comprised of the FCC complex, inertial sensors, cockpit controllers, the ISA's and LEF actuators, the AIU, and the FCS control panel. The avionics system elements used to provide the principal Pilot/Vehicle Interface (PVI) for DFCS mode control and status annunciation include the following:

- Two interactive Multipurpose Display (MPD's).
- Two Programmable Display Generators (PDG's).
- A dual redundant Stores Management Set (SMS) Control Interface Unit (CIU).
- A redundant multiplex (MUX) data bus.
- Primary and backup MUX data bus controllers.

The primary avionics MUX bus controller will be the Fire Control Computer (FCC), and the backup bus control will be provided by the SMS.

Figure 51 shows some of the avionic type components that the DFCS will interface with via the avionics and display multiplex buses. The safety of the AFTI/F-16 aircraft and the fail-safe integrity of the FCS will not be affected by failures of the dual avionics bus or any of the interfaced equipment.

The DFCS/avionics functional interface provides the following benefits:

- Data entry capability.
- Display of mode requested mode status.
- Status and fault reporting.
- Built-in test capability.

Figure 52 illustrates some of the payoffs obtained by optimal integrations of the avionics systems and the DFCS.

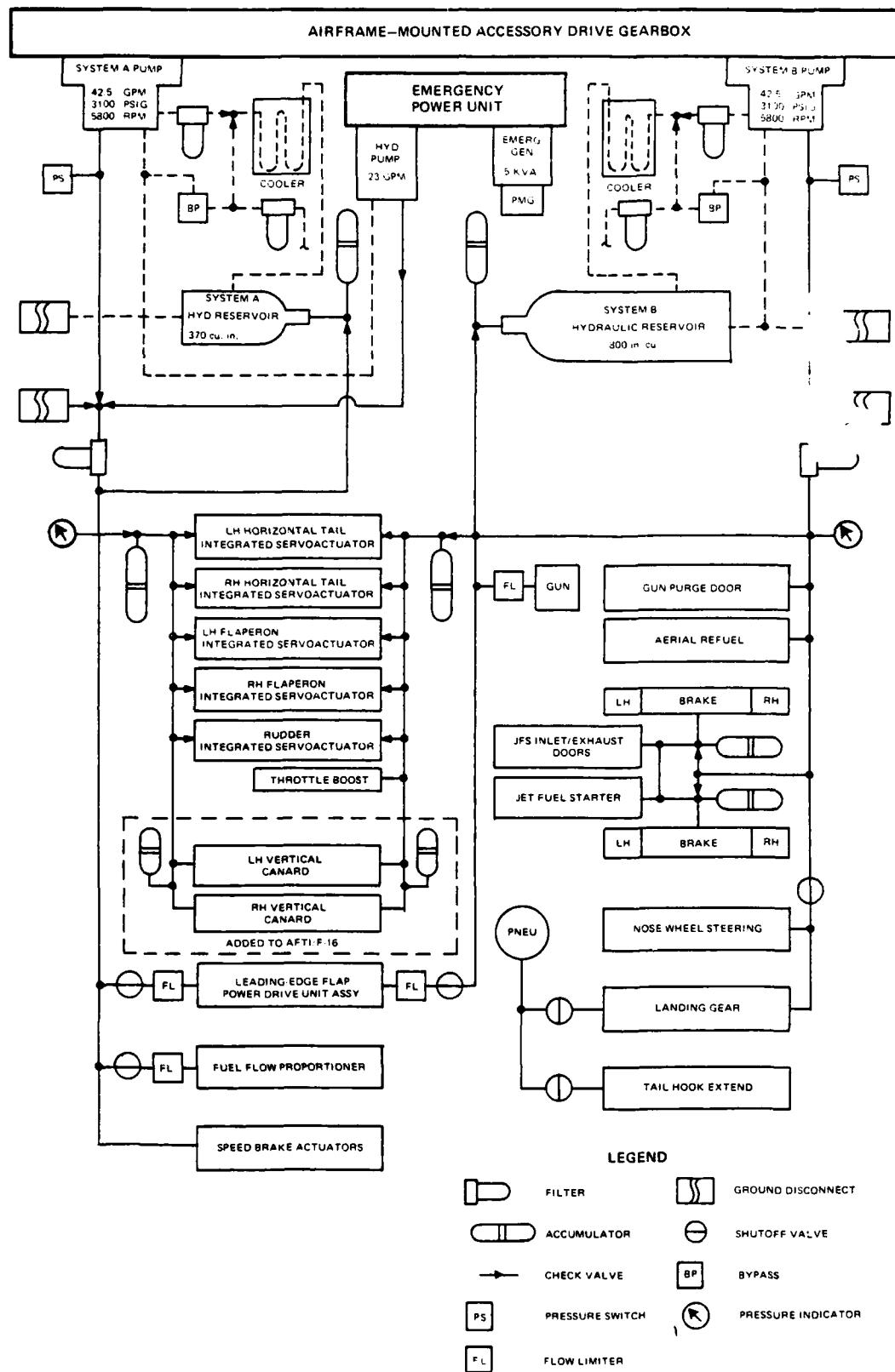


Figure 49. Hydraulic System Functional Block Diagram

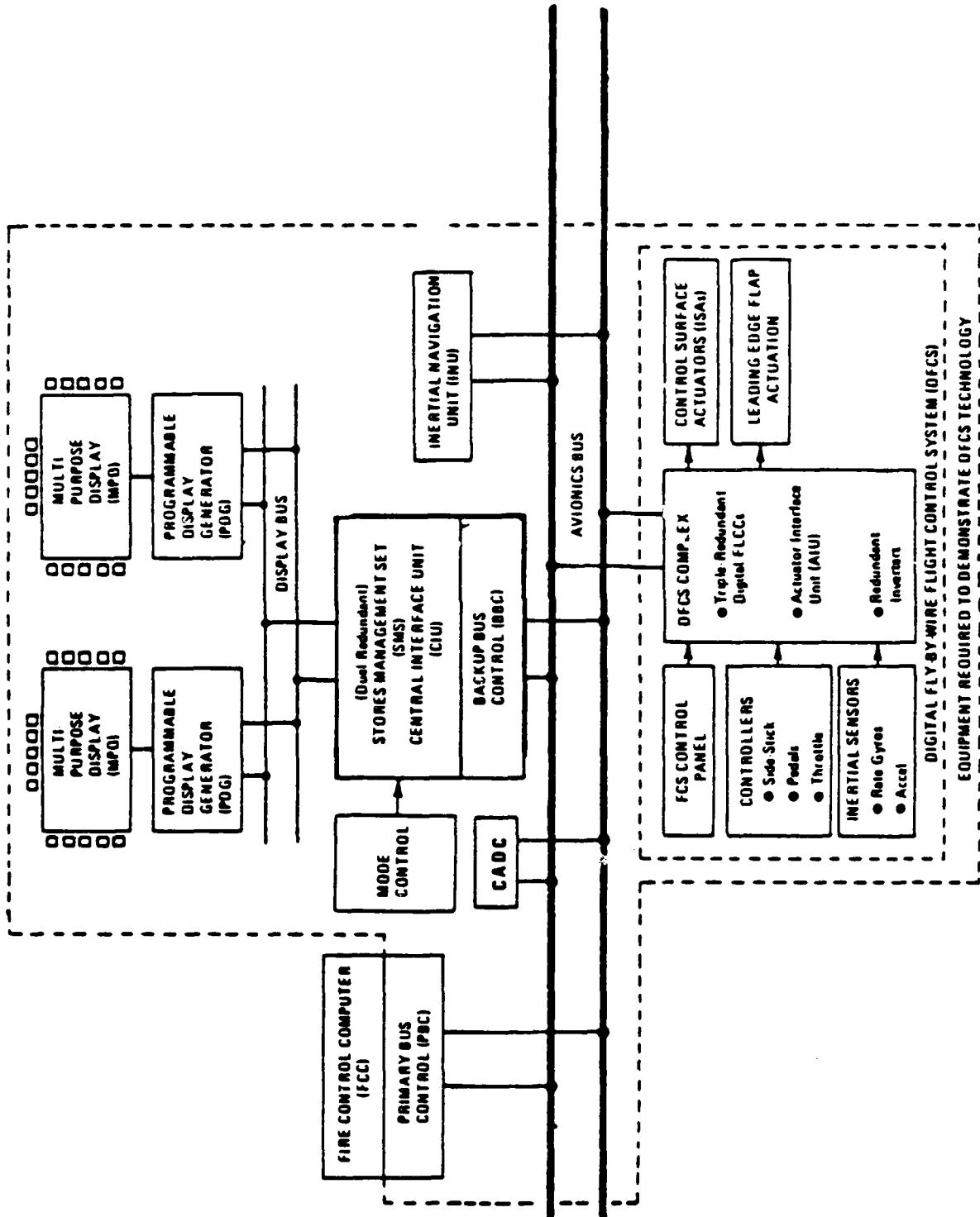


Figure 50. DFCS and Interface with Avionics

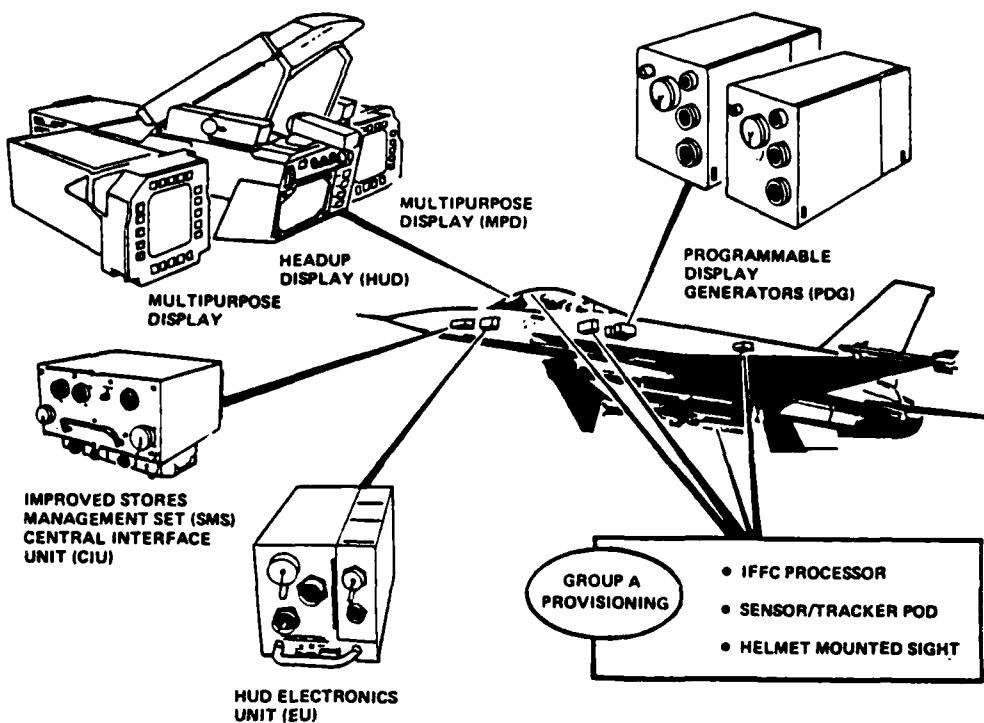


Figure 51. AFTI/F-16 Avionic Components

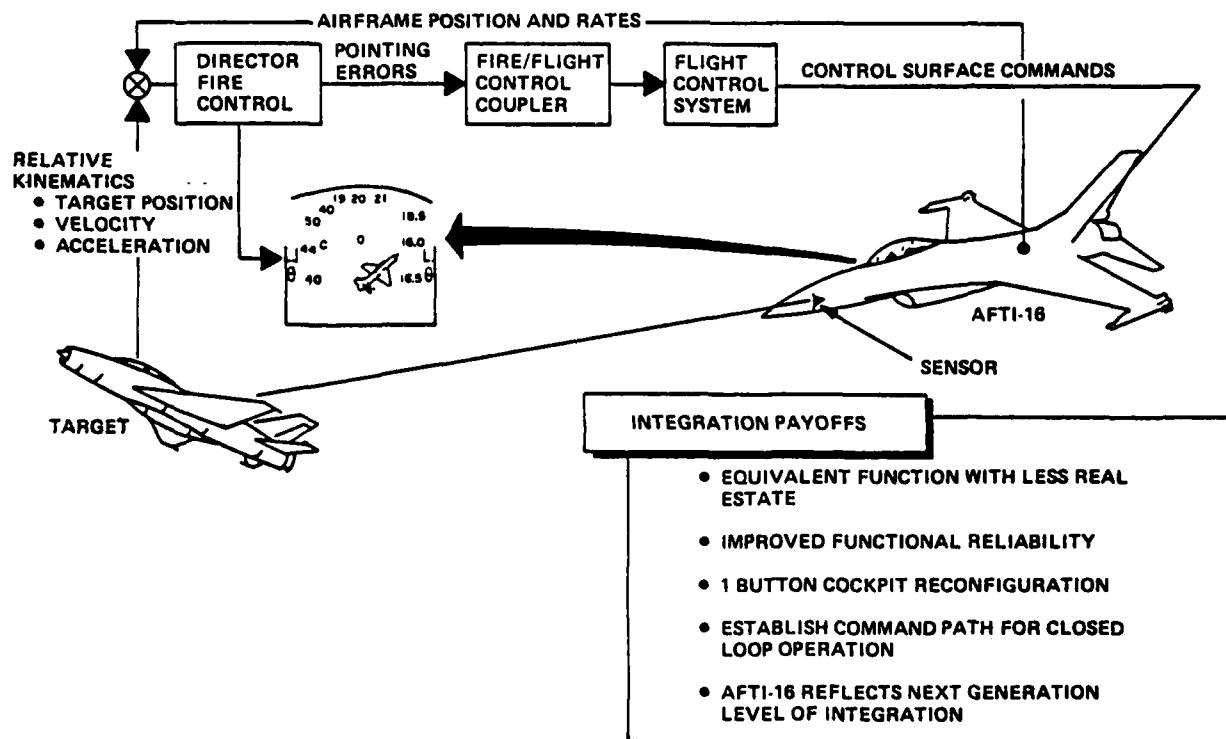


Figure 52. AFTI/F-16 Avionics/DFCS Integration Technology Payoffs

PILOT/VEHICLE INTERFACE (PVI)

The AFTI/F-16 crew station is shown in Figure 53. This arrangement was established to provide an effective Pilot/Vehicle Interface (PVI) consistent with the multimode task - tailored FCS design philosophy. Cockpit Controlled Control/Display System implementation is based on mission task requirements.

1. Cockpit Controllers

The operation of the cockpit controllers will now be reviewed. In the pitch axis the SSC always commands normal acceleration so that command ambiguity is removed. The decoupled direct lift SSC commands generate coupled normal acceleration response after the flaps reach maximum deflection. This provides the pilot with full authority control of normal acceleration. Manual vernier control of pitch pointing and vertical translation is accomplished using the left hand controller (Linear Throttle), a unique twist grip controller physically integrated with the throttle grip assembly.

The SSC lateral force commands roll rate at all times. For the standard operation, the rudder pedals command the following:

- Conventional rudder deflection with stability augmentation in the normal mode.
- Direct side force (flat turn) in all of the combat modes.

In the decoupled operations, the rudder pedals command decoupled motions that are in concert with the pitch axis control and are task oriented.

2. Control/Display System (CDS)

The AFTI/F-16 DFCS is designed to provide integrated control for changing mission segments with a single stroke of one of several keys integrated within the Cockpit/Display System (CDS). The CDS is designed to accomplish the PVI function in a manner which permits effective mode selection. The CDS provides the capability for integrating multiple equipment control functions and displays on a single panel. In addition, the CDS has the capability to automatically configure the avionics and DFCS upon selection of a particular mission phase.

The CDS consists of a Head-Up Display (HUD), two interactive Multipurpose Display (MPD), and dedicated cockpit switches (see Figure 53). The MPD's provide control/display/status functions for the avionics and the DFCS.

The CDS provides redundant DFCS/ avionics interfaces in the DFCS safety-critical paths to ensure flight safety. These interfaces are single-failure tolerant so that no one failure in the MPD's/DFCS interface will decrease the number of functions controlled through the MPD's. Display symbology and formats provide the information required by the pilot to effectively perform mission-related tasks.

Figure 54 shows the flight control options displayed on the left MPD.

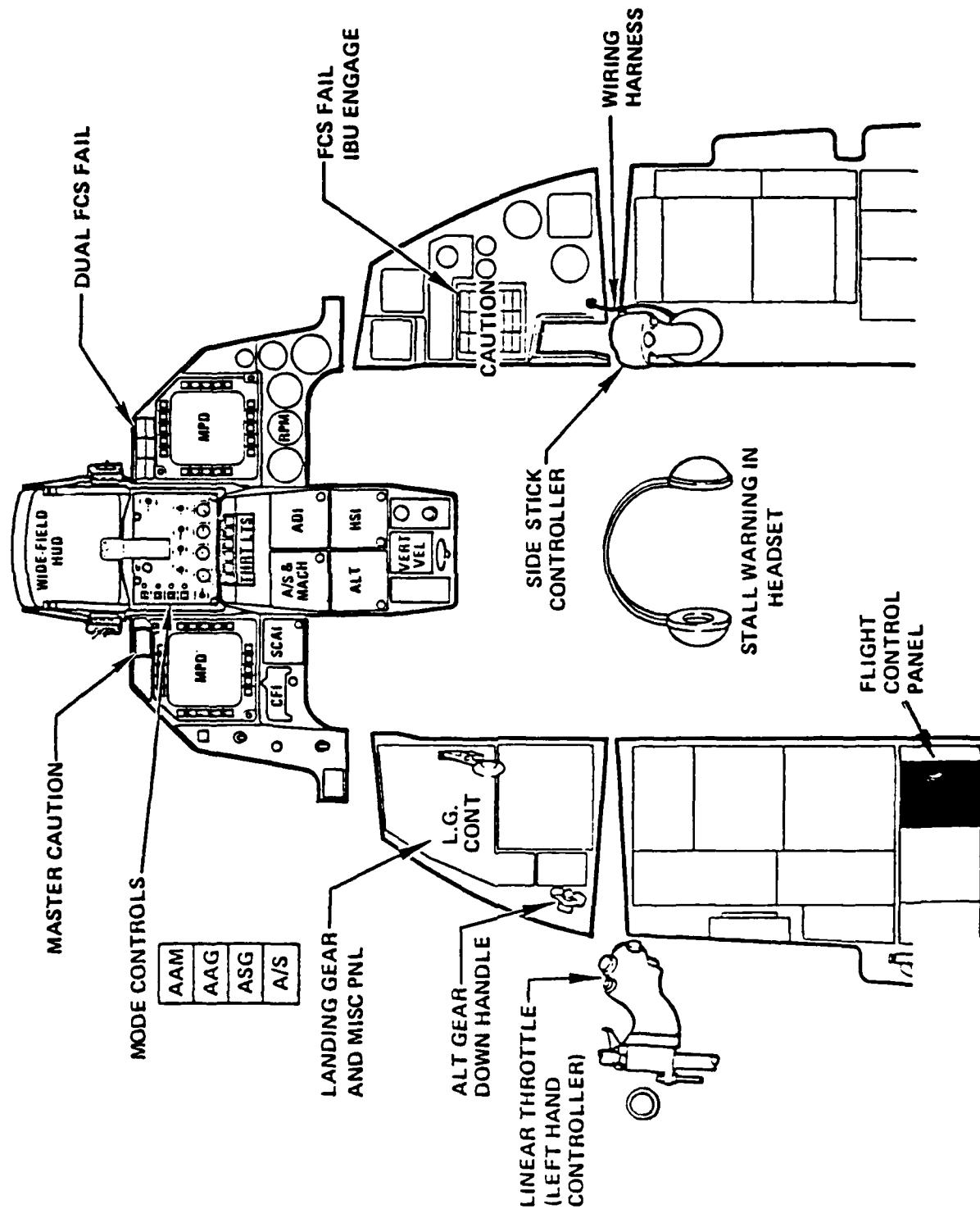


Figure 53. AFTI/F-16 Crew Station Layout

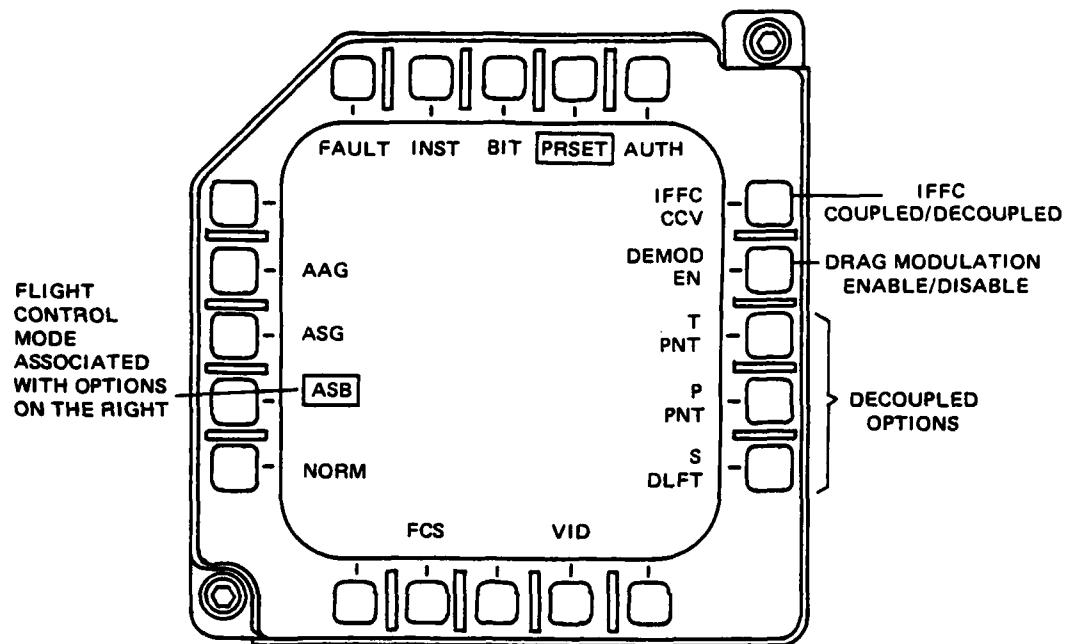


Figure 54. Flight Control Options Displayed On MPD.

From the flight control standpoint a single stroke of one of the keys shown will perform the following functions:

- Enables the MPD to present the pertinent mode information.
- Commands the FCC's to execute the control laws optimized for the mission segment.
- Enables the preselected manual 6 DOF modes on the left hand controller (throttle grip) and the rudder pedals.
- Presents the proper symbology on the Radar/EO and HUD displays.

The pilot can then engage or disengage the selected decoupled mode. The 6 DOF modes implemented may be modified at any time by pressing the appropriate switches on the MPD. Pressing the "PRSET" key will cause the display shown in Figure 54 to be presented.

By use of the MPD, the pilot can select, by flight control mode, any of the following options:

<u>CONTROLLER</u>	<u>OPTIONS</u>
• Side Stick Controller (SSC)	• Flight Path Maneuver Enhancement • Pitch Rate Maneuver Enhancement
• LH Controller (on throttle grip)	• Pitch Pointing • Vertical Translation • Direct Normal Force (Direct Lift)
• Rudder Pedals	• Direct Sideforce (flat turn) • Yaw Pointing • Lateral Translation • Blended Pointing

THE ADVANCED FLIGHT CONTROL MODES OF THE AFTI/F-16

The AFTI/F-16 is intended to provide superior flying qualities by incorporation of the ability to tailor aircraft response to a particular mission segment and by the utilization of vertical canards and maneuvering flaps to further enhance mission effectiveness. Conventional non-multimode designs generally result in a compromise between smooth ride and good handling qualities. The AFTI/F-16 FCS will be designed to void this compromise by emphasizing tracking and handling qualities during attack mission phases and ride qualities elsewhere.

1. The Four Major Flight Control Modes -

The AFTI/F-16 DFCS demonstration vehicle will be provided with four major flight control modes and decoupled CCV modes utilizing redundant control surfaces in each control axis, (Figure 55), which can be selected in any pitch and yaw combination to tailor aircraft to the mission. The major modes are as follows:

- **Normal Mode** - The normal mode is used throughout the applicable flight envelope for takeoff, cruise, and landing and for the performance of secondary mission tasks, such as air refueling and formation flying. The mode is designed to provide smooth ride, gust alleviation, and reduction of pilot workload during secondary mission segments.
- **Air-to-Air Gunnery (AAG) Mode** - The air-to-air gunnery mode is used throughout the air combat flight envelope to provide rapid maneuvering during target intercept and precise tracking. Control law design strategy for this mode is based on optimizing tracking characteristics. The pilot is then able to quickly null target tracking error while he obtains improved handling qualities by using the trailing-edge flap as a longitudinal control surface. The resultant optimized response is then applicable to a wide variety of air-to-air target acquisition and tracking tasks.
- **Air-to-Surface Gunnery (ASG) Mode** - The air-to-surface gunnery mode will provide rapid and precise pointing for increased accuracy and survivability when ground targets are being strafed. Design criteria used to null target tracking error are the same as those developed for the air-to-air mode, with additional emphasis placed on improving gust alleviation response.
- **Air-to-Surface Bombing (ASB) Mode** - The bombing flight path mode will provide precise control of the aircraft flight path and will improve gust alleviation to facilitate bombing accuracy and to enable the employment of effective control strategies in order to increase aircraft survivability.

Figures 56 thru 59 inclusive illustrate how these modes are executed by the pilot.

2. The Control Configured Vehicle (CCV) Modes.

Adjunct CCV modes preselected through the display subsystem and featuring control inputs through the twist-grip throttle and the rudder pedals will provide added capabilities. These modes include vertical and horizontal translation, pointing, and direct force. Any longitudinal and lateral combination of adjunct modes may be selected translation may be used with the mode. For example, vertical and horizontal translation may be used with the normal mode for formation flying; longitudinal pointing and lateral direct force may be used to augment the air-to-surface gunnery mode. Dominant features of the standard and decoupled DFCS multimodes are summarized separately for pitch and lateral-directional control in Figures 60 and 61.

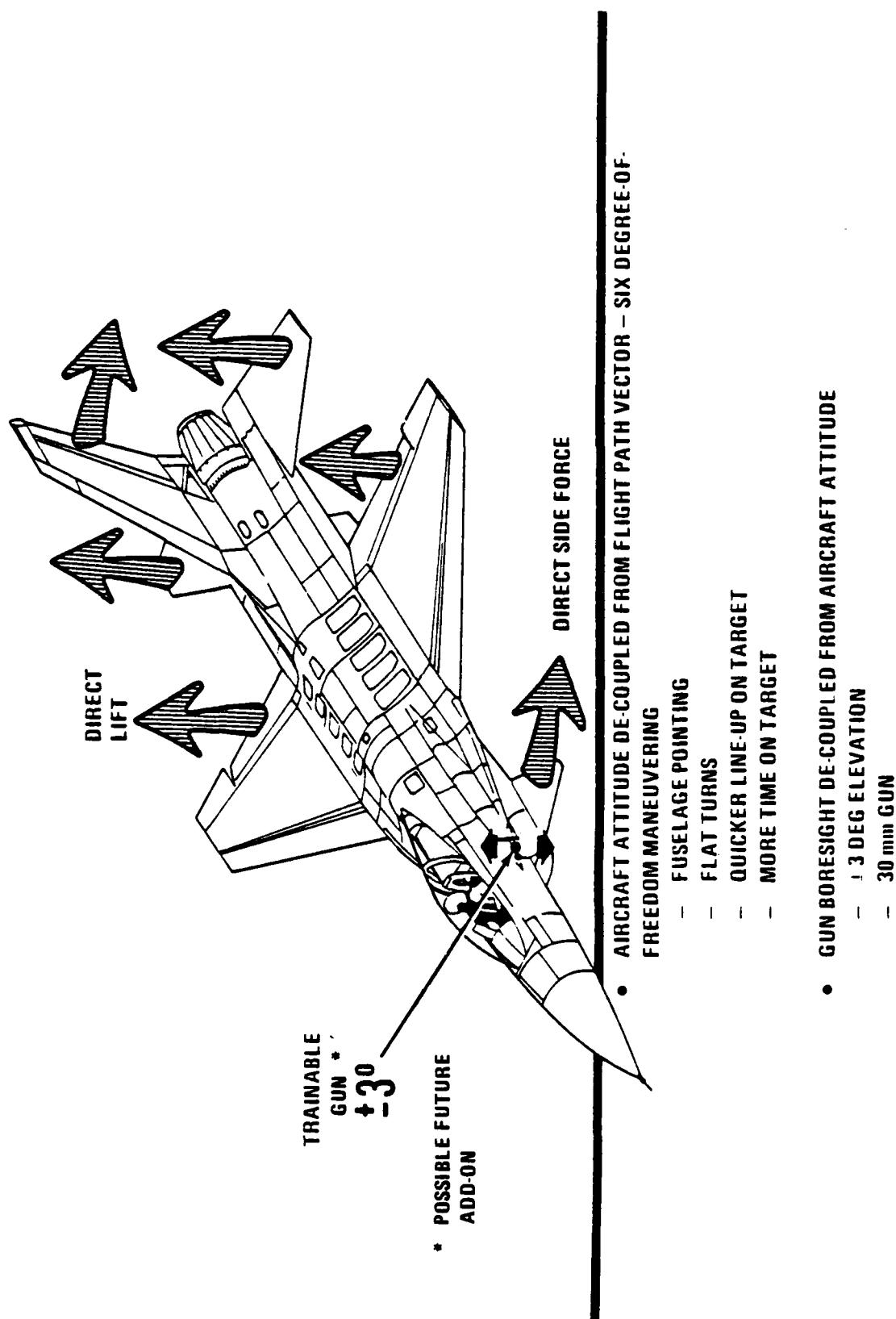


Figure 55. Control Surfaces Used for Decoupled Flight Path Control

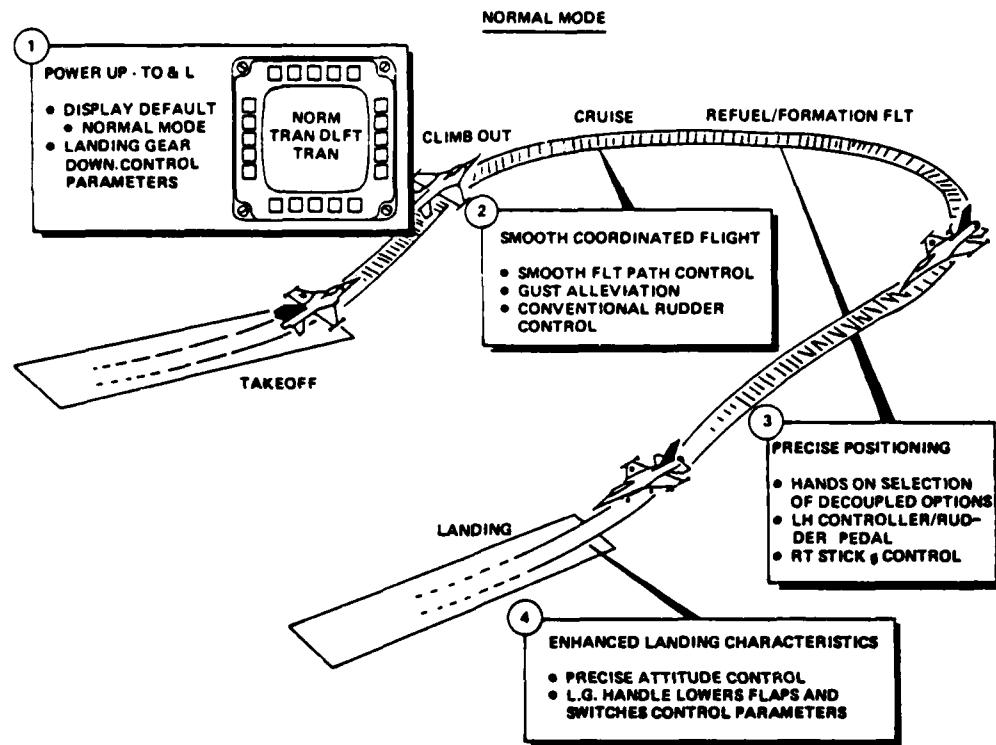


Figure 56. The Normal Mode

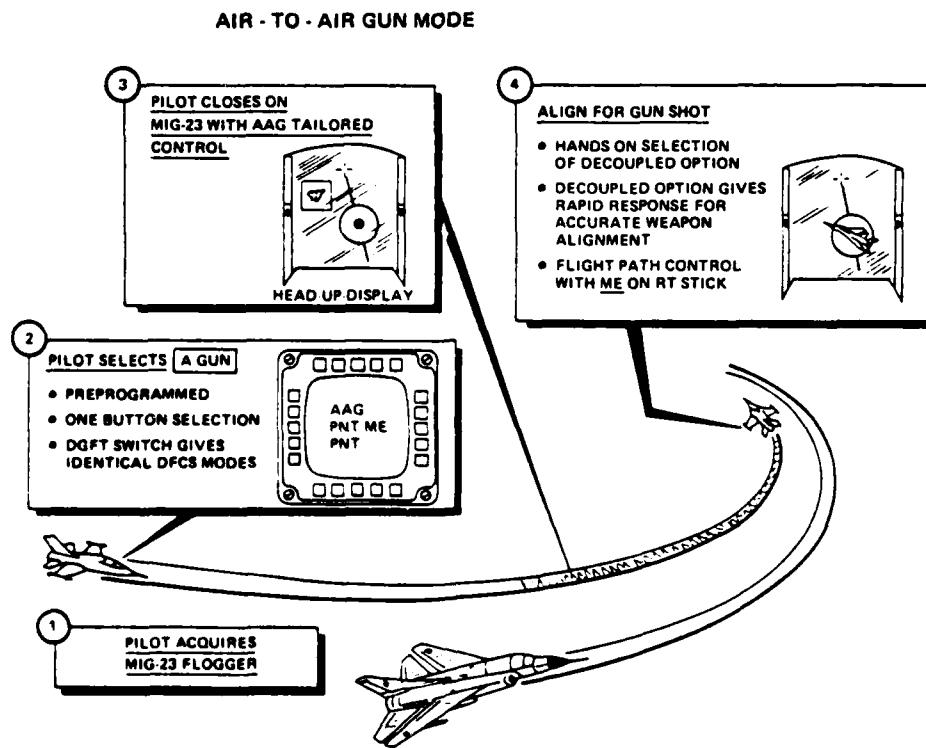


Figure 57. Air-to-Air Gunnery (AAG) Mode

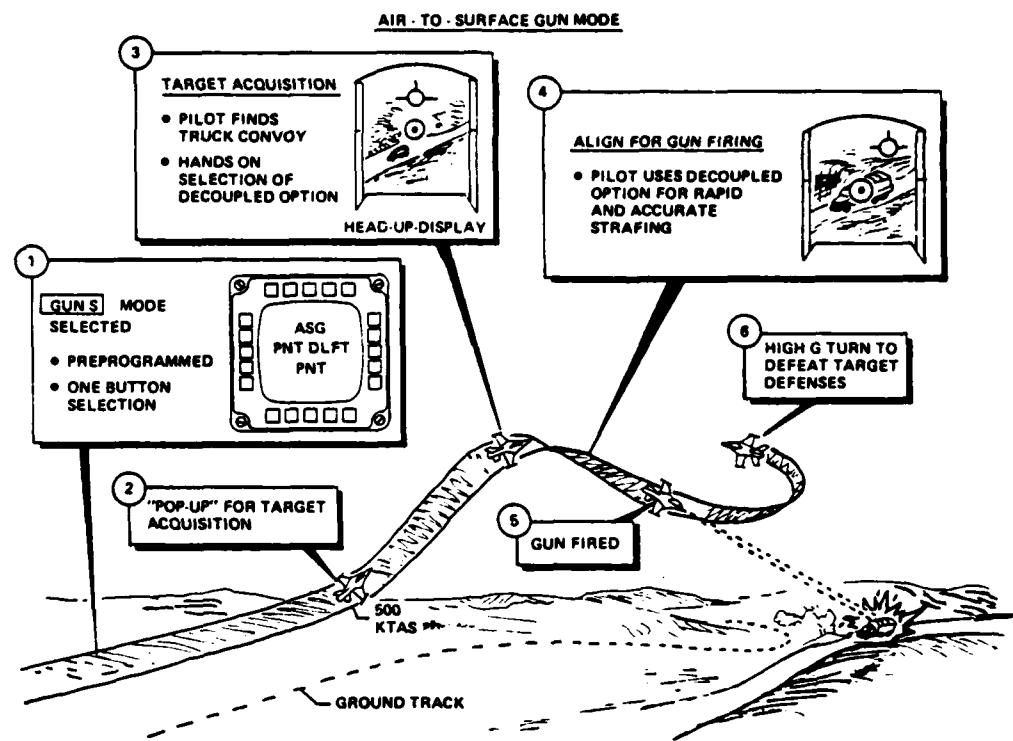


Figure 58. Air-to-Surface Gunnery (ASG) Mode.

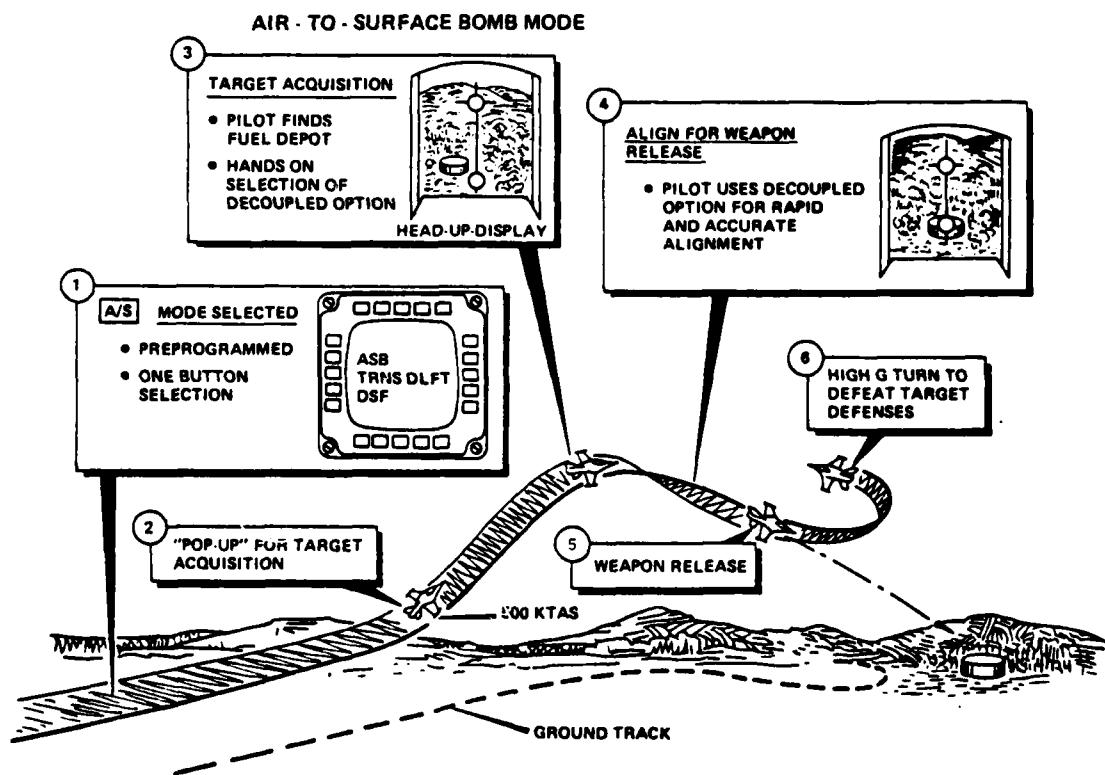


Figure 59. Air-to-Surface Bombing (ASB) Mode.

3. Control Law Design Methodology

The AFTI/F-16 multimode flight control laws are structured to provide both standard conventional control and decoupled control. The standard control characteristics are tailored for maneuvering flight and target acquisition, and the decoupled control characteristics are tailored to vernier tracking and small-amplitude-weapon delivery-error removal. High performance, unique maneuverability, rapid pitch and roll response, neutral speed stability, turn coordination, departure prevention, high angle-of-attack (AOA) performance, and high AOA command limiting are major design considerations in the development of AFTI/F-16 multimode control laws. Flying quality requirements, such as MIL-F-8785B and suggested revisions thereto, have been developed for use in the design of conventional aircraft and will be satisfied by the AFTI vehicle in the conventional modes. Since, by use of these criteria, deficiencies have been discovered in accurately predicting pilot ratings and tracking characteristics for the high-order system associated with highly augmented aircraft, additional criteria — such as C*, D*, Neal-Smith, and the newly developed step-target tracking techniques — were used to compare designs. Six-degree-of-freedom simulation results will provide the final test and acceptance criteria for all modes before flight testing.

This multimode design flexibility allows the DFCS designer to establish gains, feedbacks, and compensation parameters not only as a function of flight condition, but also as a function of the mode selected. The system performance can then be tailored to match the desired characteristics of a specific task at hand, e.g., air-to-air combat, air-to-surface bombing, etc. Besides the design flexibility obtained with a multimode system, the AFTI/F-16 vehicle utilized the trailing-edge flap as an additional control surface (along with the horizontal tail) in the longitudinal axis to enhance its performance capability. The flap used in conjunction with the horizontal tail can produce changes in the lift vector without the corresponding rotation of the aircraft, characteristic of systems using only the horizontal tail for control. The motions of both flap and tail can be coordinated by proper control system design to provide maneuver enhancement, decoupled motions, and gust alleviation to the aircraft.

The four basic task-tailored modes significantly affect the inner-loop design of the flight control system. It should be evident from examining Figures 60 and 61 that the modes can also be categorized by the influence on performance of aircraft flight path, as in the Normal and Bombing Modes, or aircraft attitude as in the Air Combat and Strafe modes. This influence is manifested in the flight control system design by controlling the normal acceleration and pitch rate responses of the aircraft, respectively.

The analytical method used to derive the longitudinal axis feedback gains for each of the AFTI/F-16 standard modes is based on optimal control theory and is referred to as linear quadratic synthesis (LQS). In this procedure, weighting parameters in a quadratic cost function are selected to yield systems with desirable performance characteristics. Weighting parameters reflect consideration of gust response, tracking performance, phase and gain margin, short period frequency and damping, transient response characteristics and other relevant performance indices. Simplified longitudinal and lateral-directional block diagrams of the Normal Mode flight control system are shown in Figures 62 and 63, respectively. In addition to the feedback gain variables, further response tailoring is provided to the system through the command path parameters, pilot gain (K_p) and pilot time constant ($1/a_p$). Optimum L/D flap scheduling is also provided.

Most of the inner loop gains are scheduled with air data derived quantities to provide good response over the entire flight envelope. In addition, some filter time constants and break frequencies are also scheduled. Structural limiting is also provided to prevent overstressing the airplane during maneuvers. One of several ways this is done is by limiting flap travel as function of air data, Mach, altitude, and load factor.

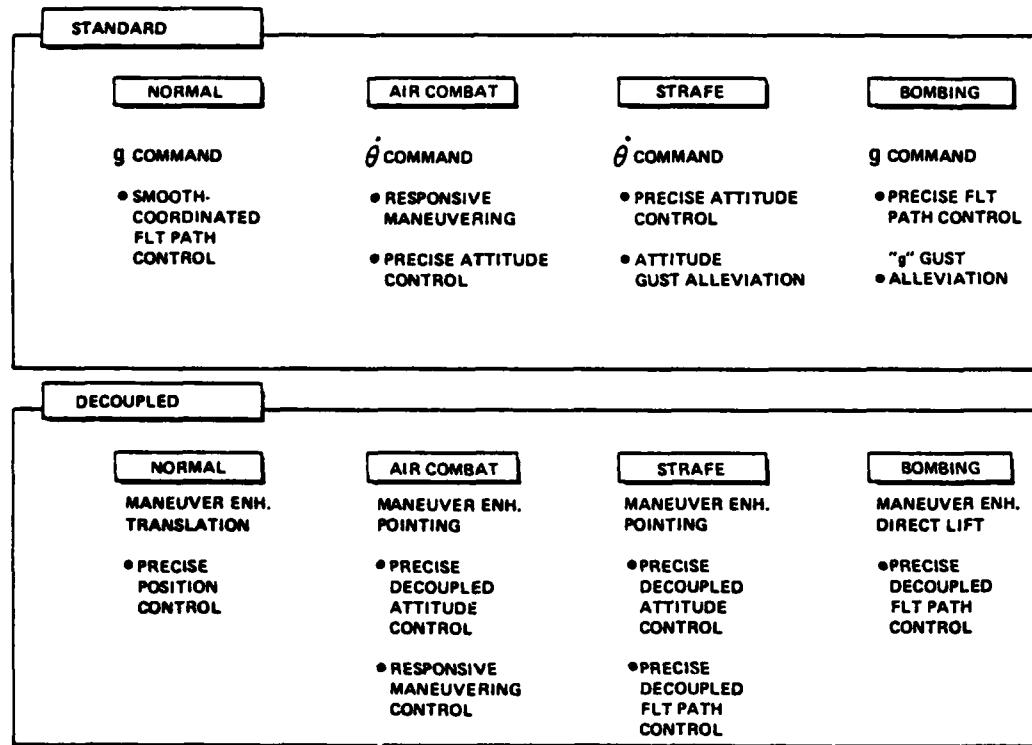


Figure 60. Pitch DFCS' Multimode Control

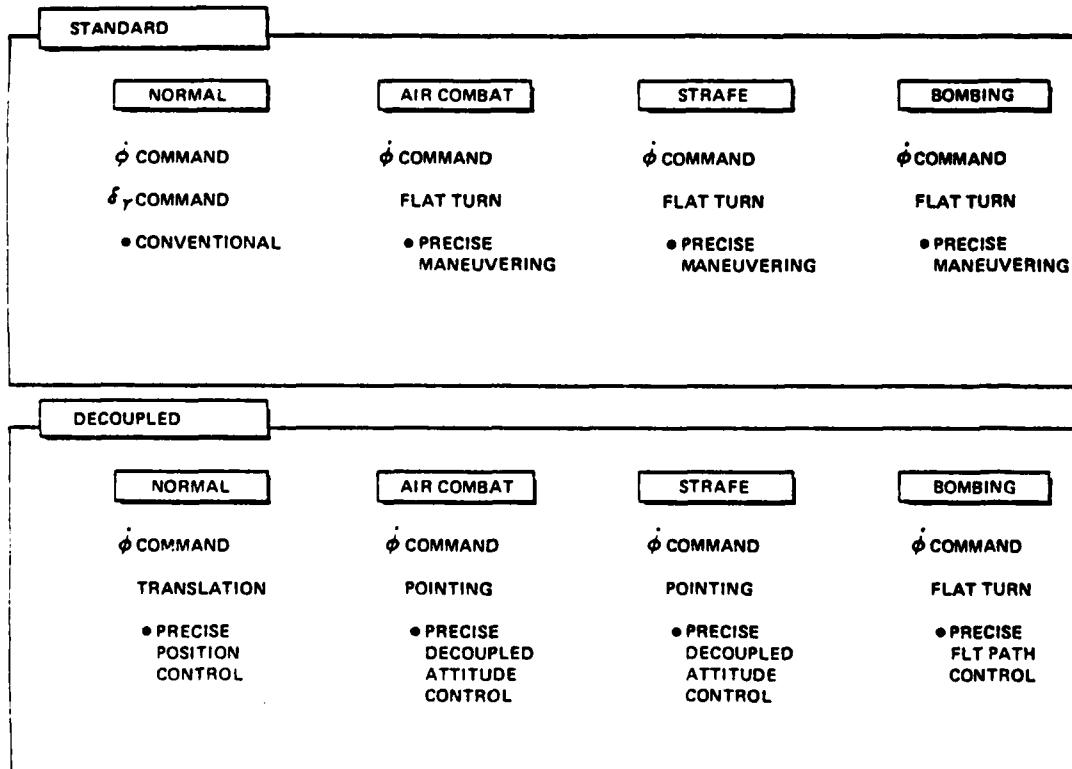


Figure 61. Lateral-Directional DFCS Multimode Control

A conventional approach to directional augmentation was selected for the system concept. Lateral acceleration feedback provides coordination; washed-out yaw-rate feedback enhances dutch-roll damping, and an aileron-rudder-interconnect (ARI) and a roll rate-AOA ($p\alpha$) interconnect improves turn coordination. The principle alternative to this design philosophy is the beta-dot (sideslip rate) feedback concept employed in the decoupled system. The beta-dot signal is synthesized using non-redundant Euler attitude and velocity data. A technique has been developed to give fail-operate performance in the decoupled control mode in the event of the loss of these data. However, higher reliability is desired for the primary takeoff-and-directional control system so a separate, more conventional, directional control system has been developed for the Standard Normal Mode. In summary, the approach that was taken in the flight control system design of the Standard Normal Mode was to develop a system that produced smooth, uniform, responsive handling qualities and provided gust alleviation of normal acceleration, while simultaneously meeting the stability and root location specification, and not violating any physical constraints. Using this design procedure, gain schedules were established over a suitable range of flight conditions.

As in the Normal Mode, certain tradeoffs arise in the development of the flight control laws in the Air Combat Mode. However, since tracking characteristics and aircraft attitude control are of primary interest in this mode, a different set of performance measures are emphasized in evaluating system designs. To aid in achieving the desired level of performance in the mode, additional compensation was added to the command path of the flight control system. The structure of this compensation is depicted in Figure 64. Because of task similarities, the Air-to-Surface gunnery mode is virtually identical to the Air-to-Air gunnery mode in the pitch axis. It should be noted that the gains shown in the block diagram will also have values different than in the Normal Mode. The methodology described above was also used in developing the Air-to-Surface bombing mode which emphasizes precise flight path control and is illustrated in Figure 65.

4. Decoupled Control Modes —

The decoupled control laws provide independent control of all six degrees-of-freedom of motion. They are accessible from all major modes. Certain default or preselected decoupled motions have been defined for the major modes but can be overridden by pilot input through the MPD's. AFTI/F-16 decoupled maneuvers are illustrated in Figures 66 thru 71 inclusive.

The control laws for implementing these decoupled modes were synthesized using classical fixed-flight condition feedback control methods. Within this classical approach, there were several ways to decouple pitch rotation from normal acceleration. One such way is to analytically derive the precise compensators, as a function of aircraft aerodynamic parameters, required to cancel the basic aircraft dynamics while at the same time substituting a set of desired dynamics for the total closed-loop system. Results of analysis showed that responses obtained from a system decoupled in this manner were unacceptably sensitive to errors in the designer's knowledge of the fundamental aircraft aerodynamic parameters.

Another approach, and the one selected for implementation, is the use of feedbacks of the variables to be decoupled to produce high-gained error signals to drive the control surfaces. This approach does not rely on unreasonably accurate estimates of the aerodynamic parameters to accomplish an acceptable degree of decoupling. When the Normal or Bombing Modes are selected, the system configuration will emphasize flight-path control whereas the gunnery modes will emphasize attitude control. An important consideration in the final selection of the decoupled variables is the fact that the IFFC algorithms produce normal and lateral acceleration commands and pitch rate, yaw rate and roll rate commands. Thus, A_n and q and A_y , β , and p were selected to be decoupled for the longitudinal and lateral modes, respectively. Coordinated control-surface limiting (CCSL) was developed

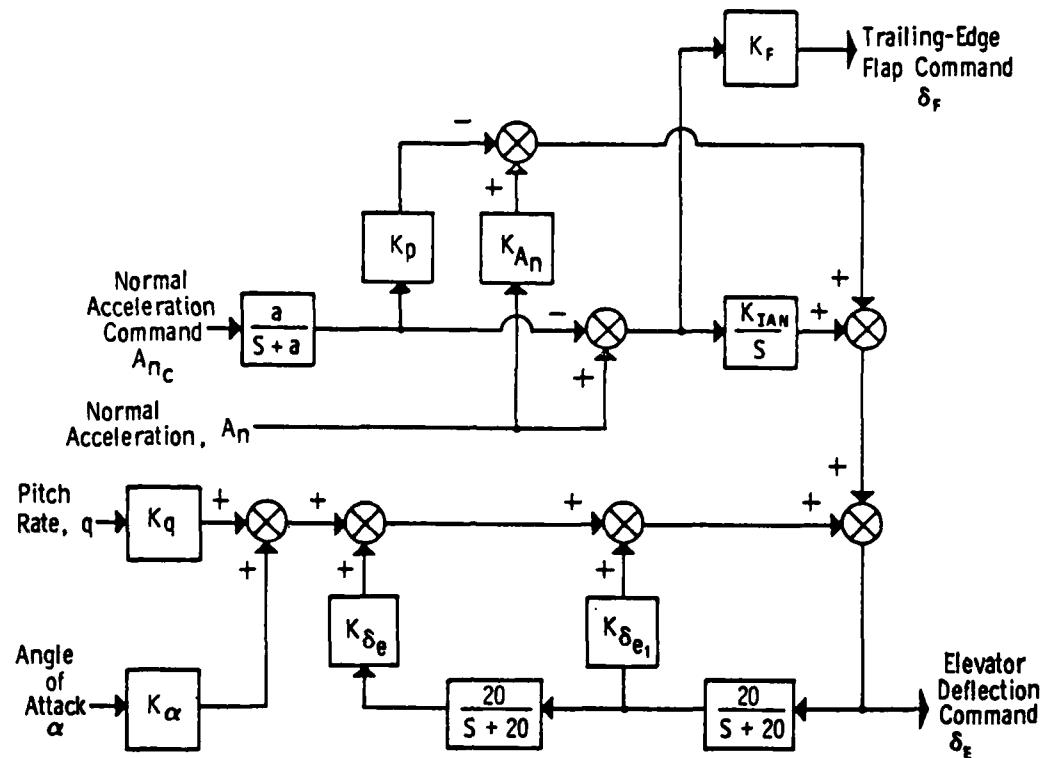


Figure 62. Simplified Longitudinal Standard Normal Mode Block Diagram

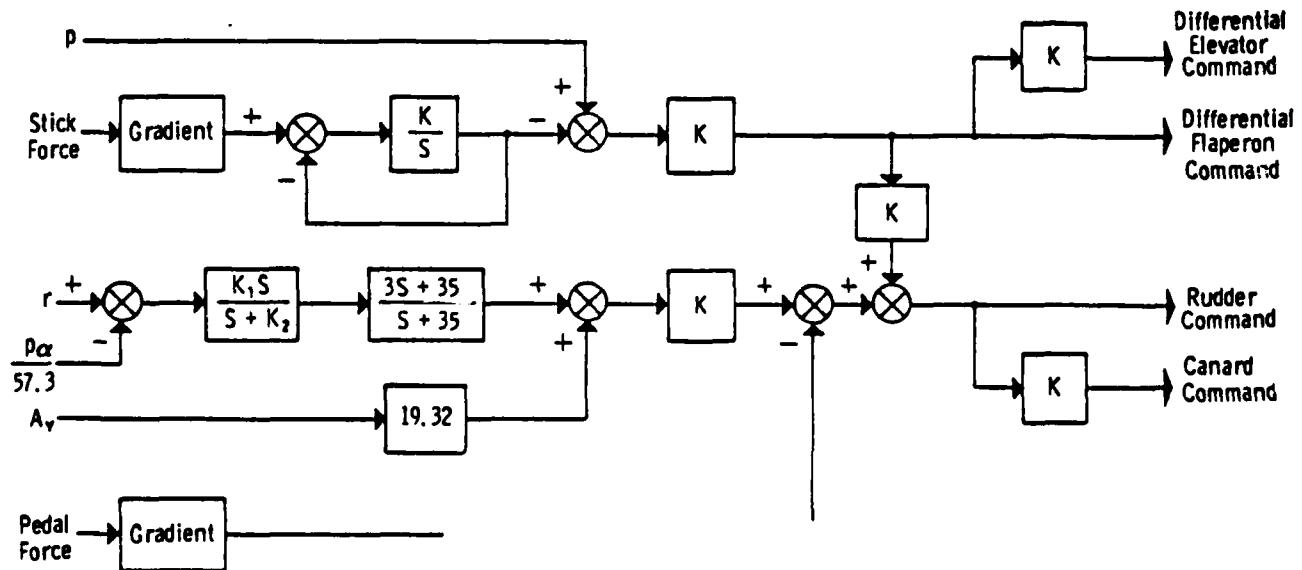


Figure 63. Simplified Lateral-Directional Standard Normal Control Mode Block Diagram.

as a means to accomplish smooth transient-free control-surface coordination and limiting. Proper decoupled mode operation is always dependent on balanced forces and moments. This requirement dictates that when a control-surface limit occurs, further decoupling inputs to other control surfaces must be inhibited and the system status maintained at that point until surface limits are no longer exceeded. This CCSL concept has been implemented, as required, in the longitudinal and lateral axes to achieve control-surface limiting while retaining the maximum available decoupled response. The engagement of the decoupled modes with correct initial conditions is accomplished in an initialization routine that automatically balances the system with the existing set of inputs and outputs.

Tables 7 and 8 list the Longitudinal and the Lateral-Direction Control Modes respectively. The controllers and control surfaces used are also listed.

Table 7
AFTI/F-16 DFCS DECOUPLED
LONGITUDINAL CONTROL MODES

- Modes
 1. ΔN – Direct Lift
 2. α_1 – Fuselage Pointing
 3. α_2 – Vertical Translation
- Controller
 - Twistable Throttle Grip
- Control Surfaces
 - Flaperons Balanced with Horizontal Tail

Table 8
AFTI/F-16 DFCS DECOUPLED LATERAL-
DIRECTIONAL CONTROL MODES

- Modes
 1. Δy – Direct Sideforce
 2. β_1 – Fuselage Pointing
 3. β_2 – Lateral Translation
- Controller
 - Rudder Pedals
- Control Surfaces
 - Vertical Canards Balanced with Rudder

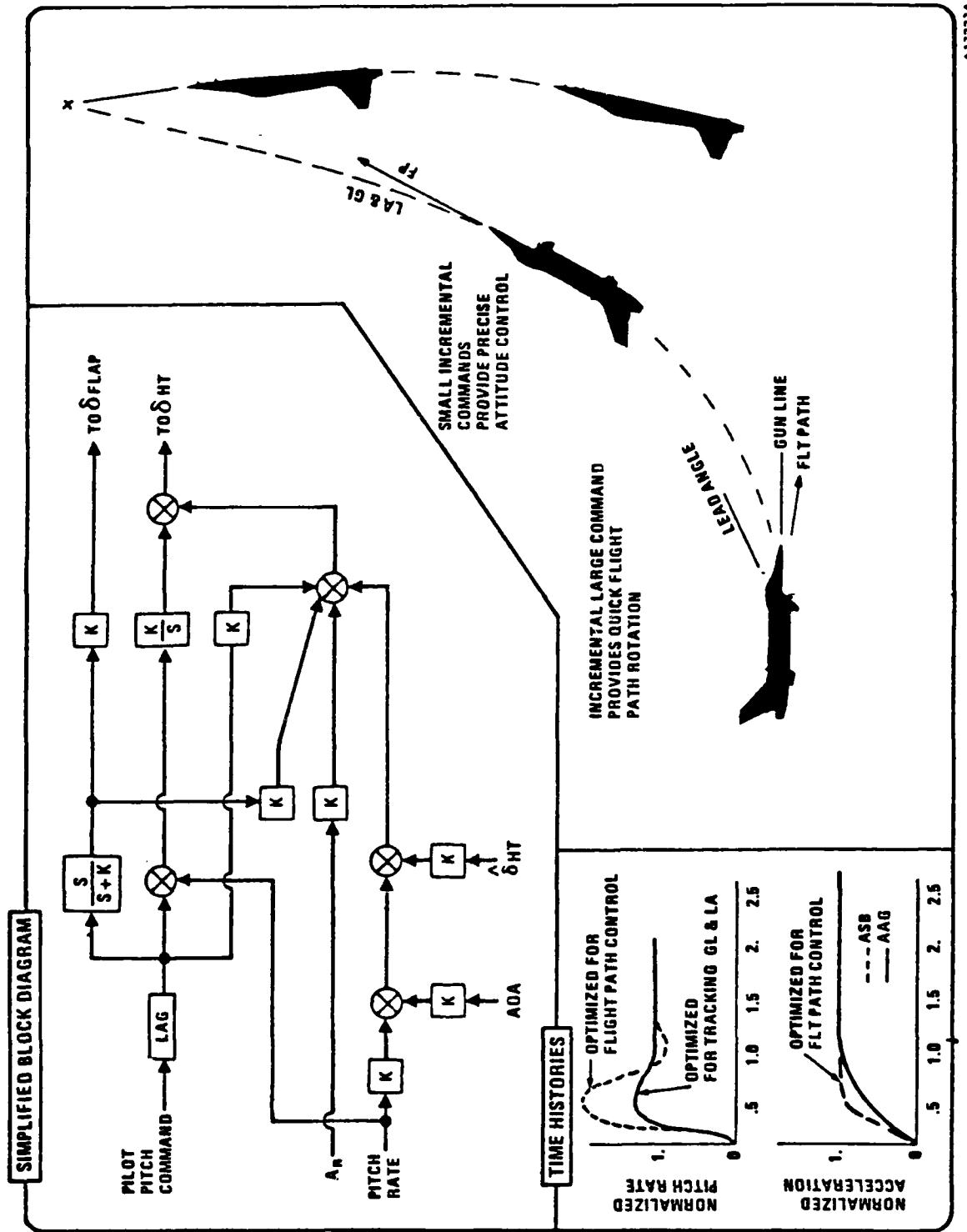


Figure 64. Simplified Air Combat Gunnery Mode.

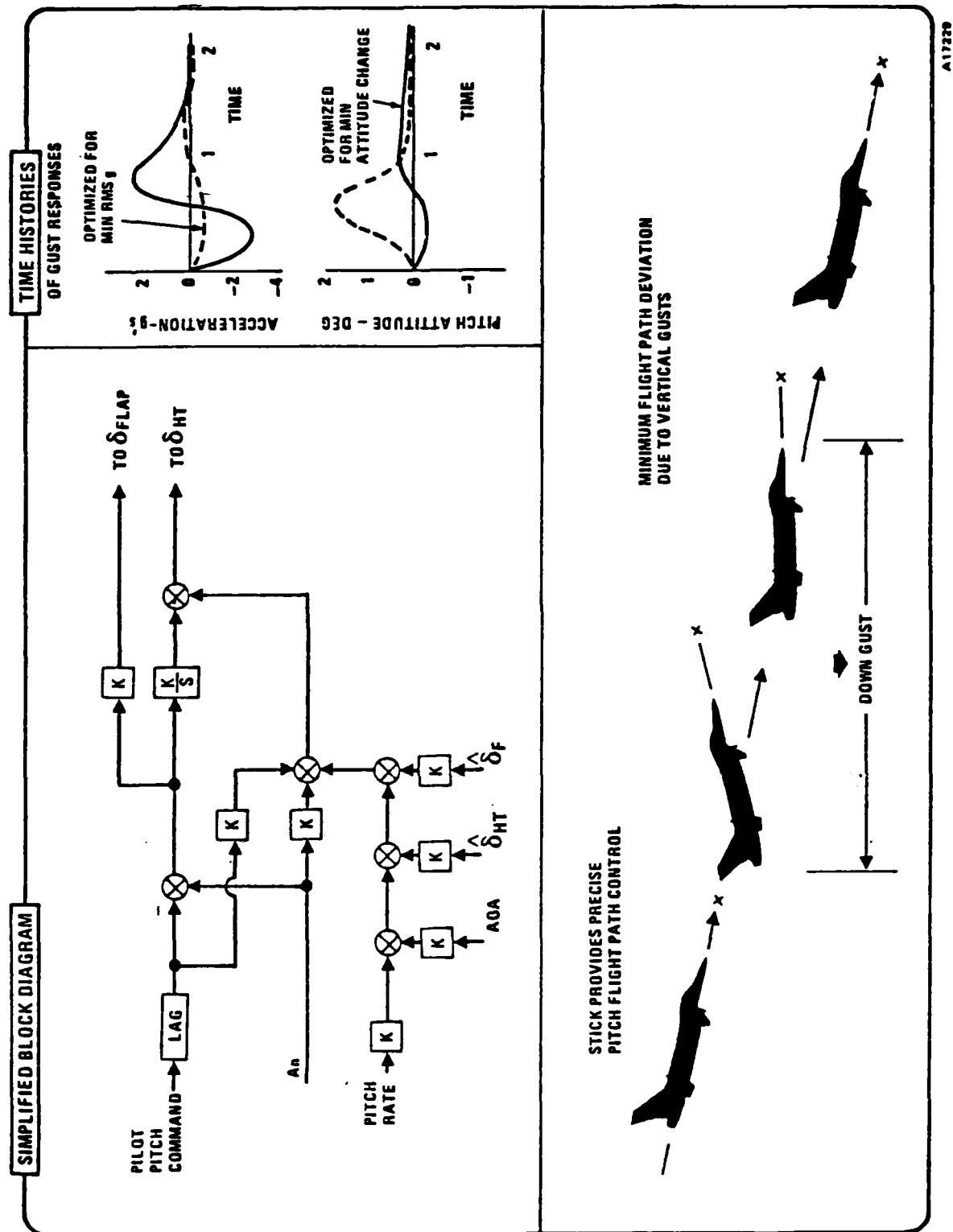
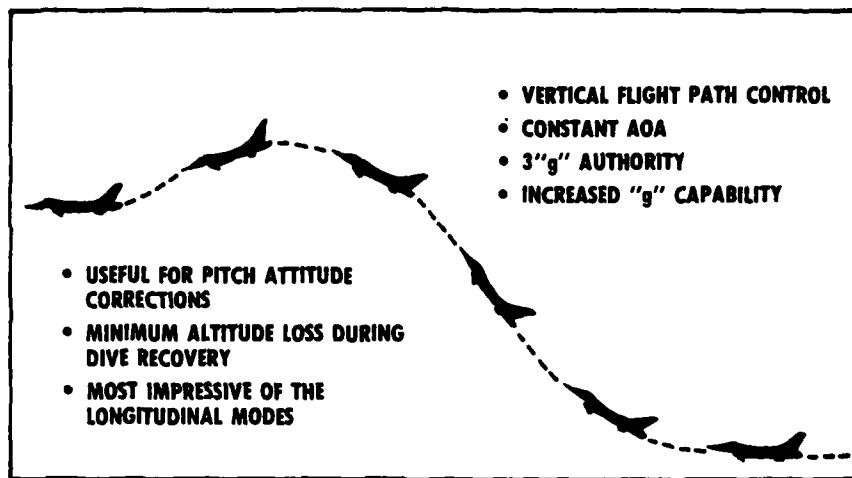
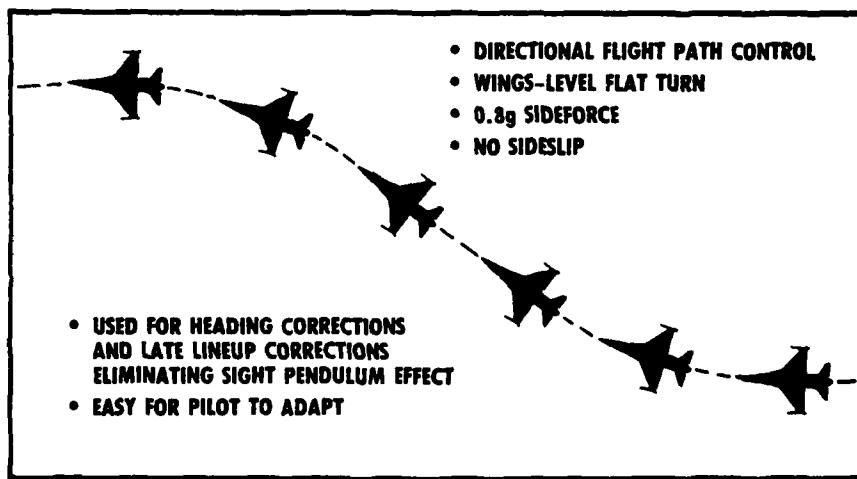
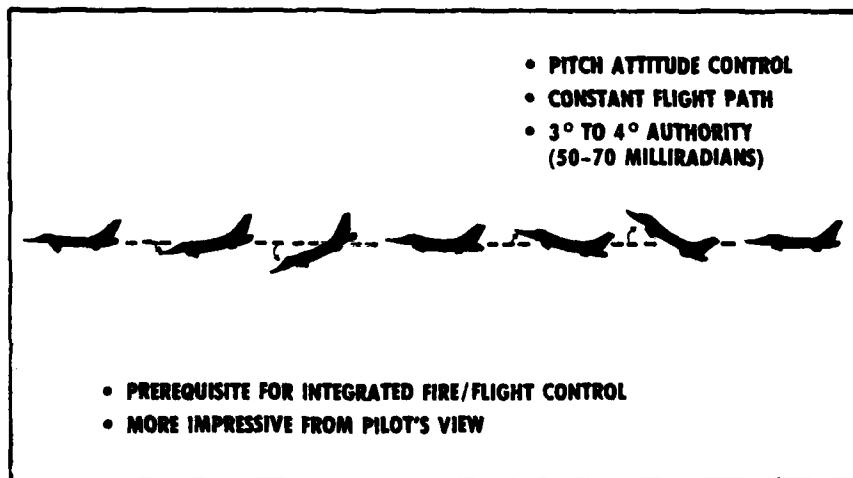


Figure 65. Simplified Air-to-Surface Bombing Mode.

Figure 66. Direct Lift (A_N) Control Mode.Figure 67. Direct Sideforce (A_y) Control Mode.Figure 68. Pitch Pointing (α_1) Control Mode

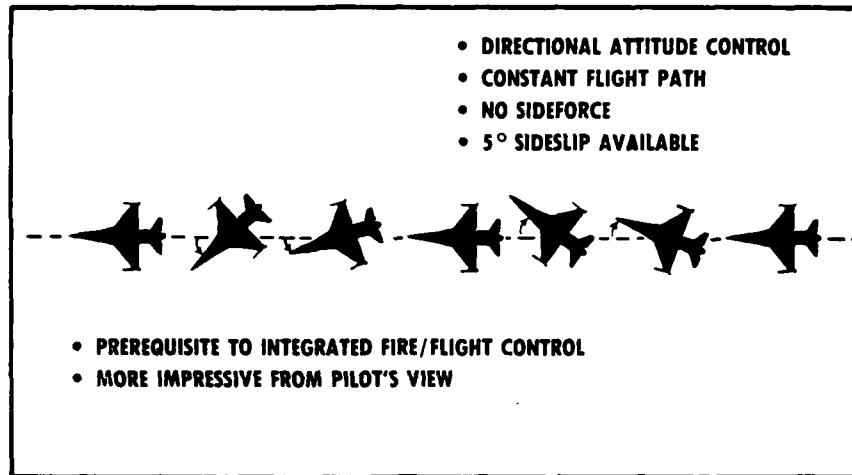
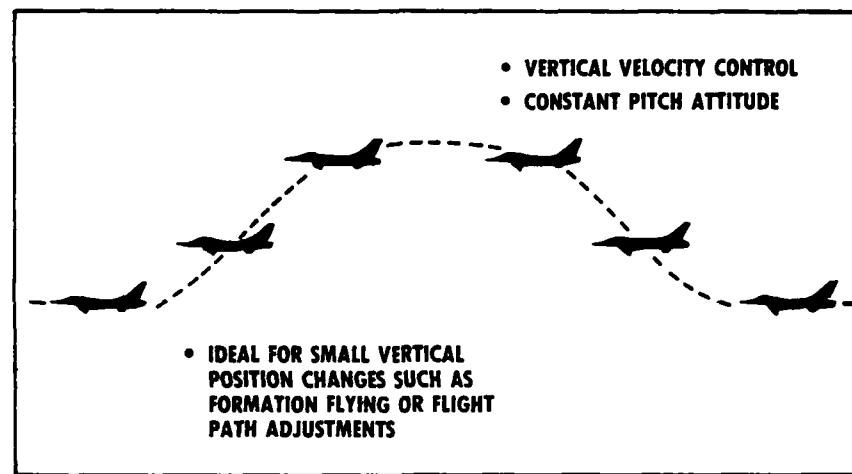
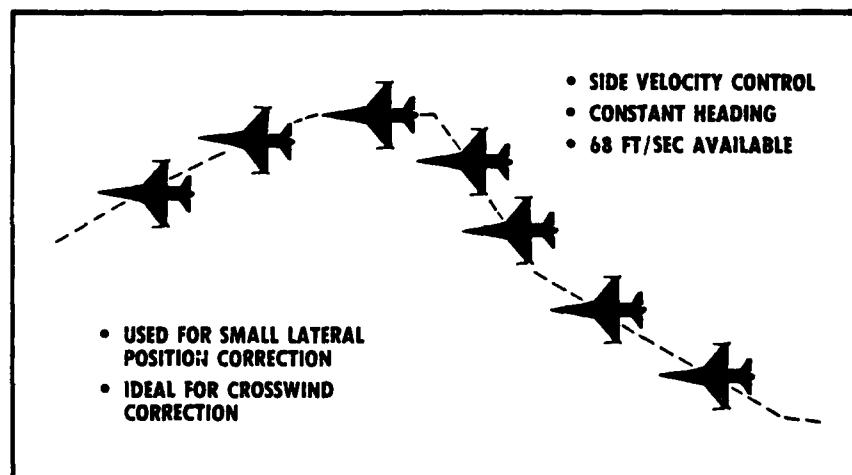
Figure 69. Yaw Pointing (β_1) Control Mode.Figure 70. Vertical Translation (α_2) Control Mode.Figure 71. Lateral Translation (β_2) Control Mode.

Figure 72 thru 74 inclusive illustrate how the use of these decoupled modes improve the operational capability of an aircraft.

a. Longitudinal Decoupling -

Since normal acceleration and pitch rate are the variables to be decoupled, these sensed quantities are fed back as shown in the Air Combat mode example of Figure 75. The sidestick controller provides direct lift commands and the throttle grip provides pointing and translation commands. A third feedback is angle of attack. Since the airframe is statically unstable over most of the subsonic flight envelope, a combination of angle of attack and pitch control law decouples this new "pre-stabilized" coupled system. Decoupling relies on the symmetrical deflection of the trailing-edge flaps as well as the symmetrical deflections of the horizontal tail. An important limiting feature is the angle-of-attack limiting. Pilot command limiting is also provided. The design also includes a flap "washout" function. To prevent the flap from achieving steady-state offset from its optimum flap schedule, the flap command is fed back in a manner which tends to return the flap to the flap position prescribed by the optimum flap schedule. The flap remains at (or returns to) the position dictated by the optimum flap schedule during relatively quiescent periods of controller activity.

Another important functional element of the system is the surface command crossfeeds. The acceleration error, which becomes ultimately the primary flap command, is also scaled by a different value and sent to the horizontal tail. The scale factor applied is theoretically the one required to alter the tail position by the amount required to cancel the change in pitching moment due to the new flap position. Conversely, the pitch-rate error is crossfed through an appropriate scale factor to the flap to cancel the change in normal acceleration created by the new tail position. The crossfeed of normal acceleration to the tail is an influential path, because even though the flap is primarily a lift-producing surface, it is also a non-negligible moment producer. On the other hand, the pitch-rate error-to-flap crossfeed is quite insignificant at most flight conditions because tail position changes produce relatively small lift changes.

b. Lateral-Directional Decoupling

To generate lateral-directional decoupled motions, the control system must be able to constrain the value of any two of the three directional states (sideforce, sideslip, heading) in addition to minimizing roll rate. In order to achieve this capability three basic design concepts are employed. The Air-to-Surface bombing flat turn mode of Figure 76 is an example. (a) A dedicated feedback loop is designed for each lateral-direction state to be controlled. Roll rate is fed back to the aileron controls; beta-dot is fed back to the rudder, and sideforce is fed back to the canards. (b) In each of the control loops, a proportional-plus-integral network is engaged to force steady state command errors to zero. (c) The underlying mathematical theory of the decoupling concept assumes that the commands from the dedicated control loops can produce pure forces or moments. For example, the roll loop command to the ailerons (flaperons) would produce a rolling moment without attendant yawing moment or sideforce. Similarly, the force-loop command would produce surface deflections do produce sideforce, rolling moment, and yawing moments, interconnect command signals are generated in order to cancel the undesirable effects. In theory, a total of 11 interconnect signals of this sort would be required to cancel the cross-coupling between system states. However, all but four of these signals can be neglected and satisfactory decoupling performance of the Standard Normal system; the roll-rate-error integrator is engaged either during gunfiring operations or when a directional command is input via the pedals. There are also two decoupling signals added to the aileron command. One signal is the canard-to-aileron interconnect; the other cancels the rolling moment induced by the sideslip which is generated during pointing or translational maneuvers.

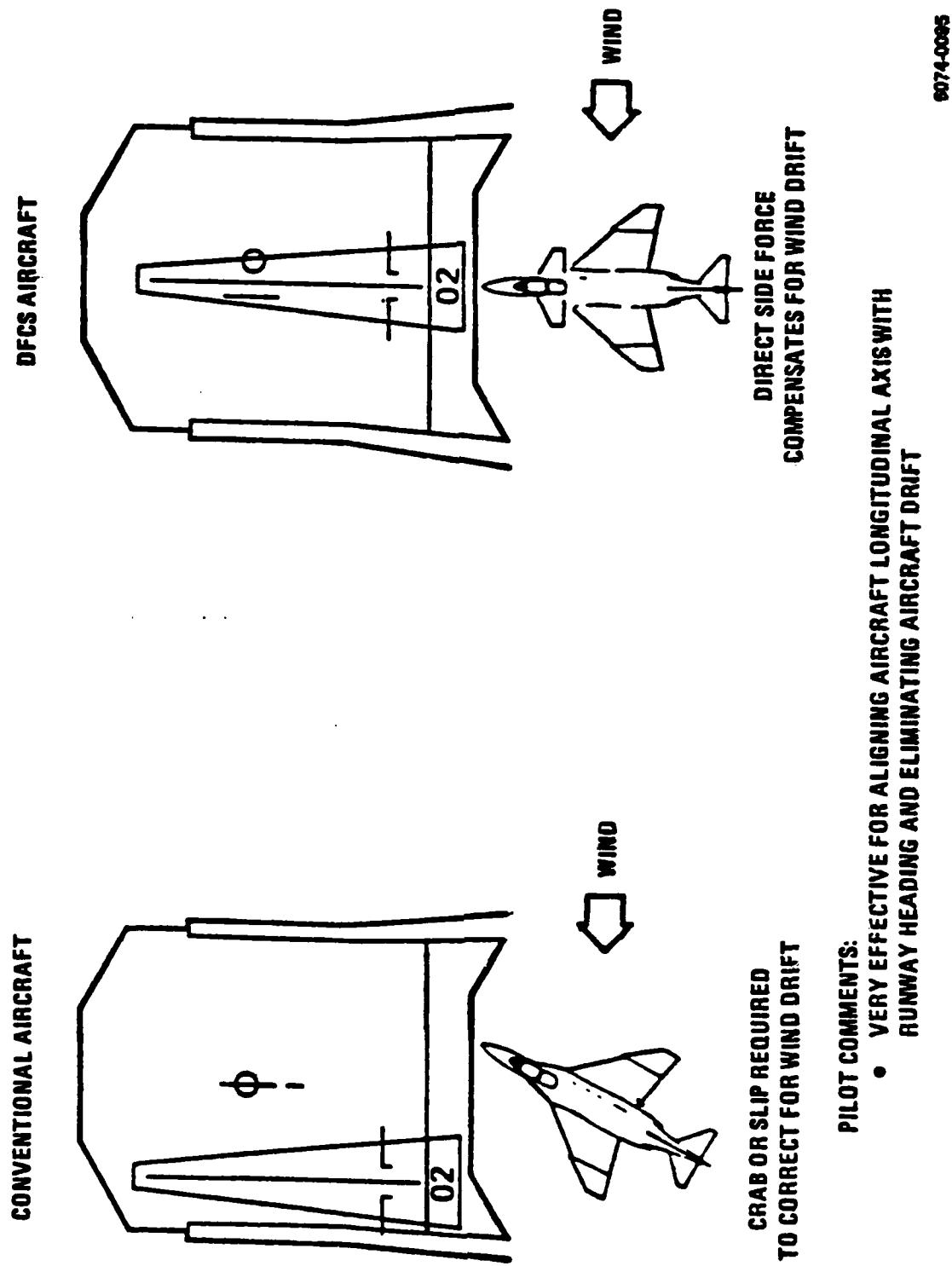


Figure 72. Effect of Lateral Translation Capability on Landing.

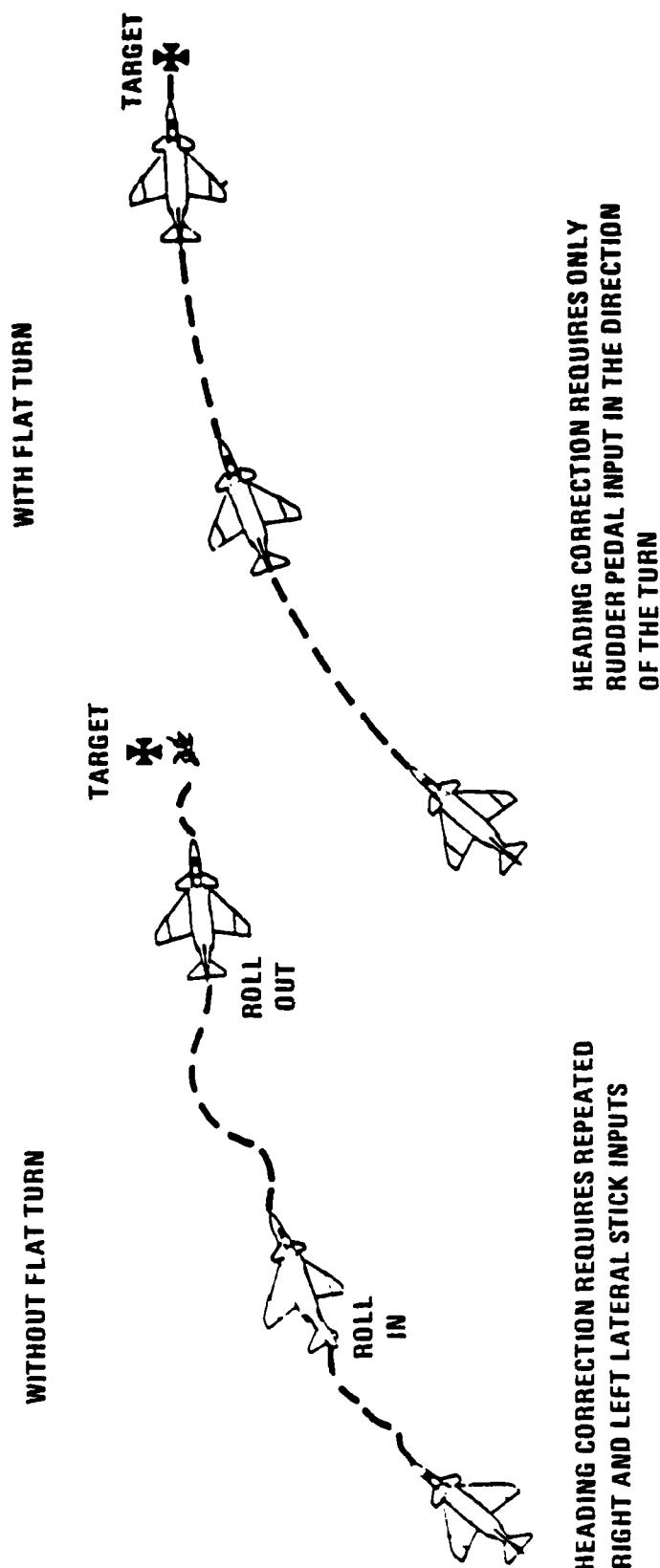


Figure 73. Effect of Flat Turn Capability.

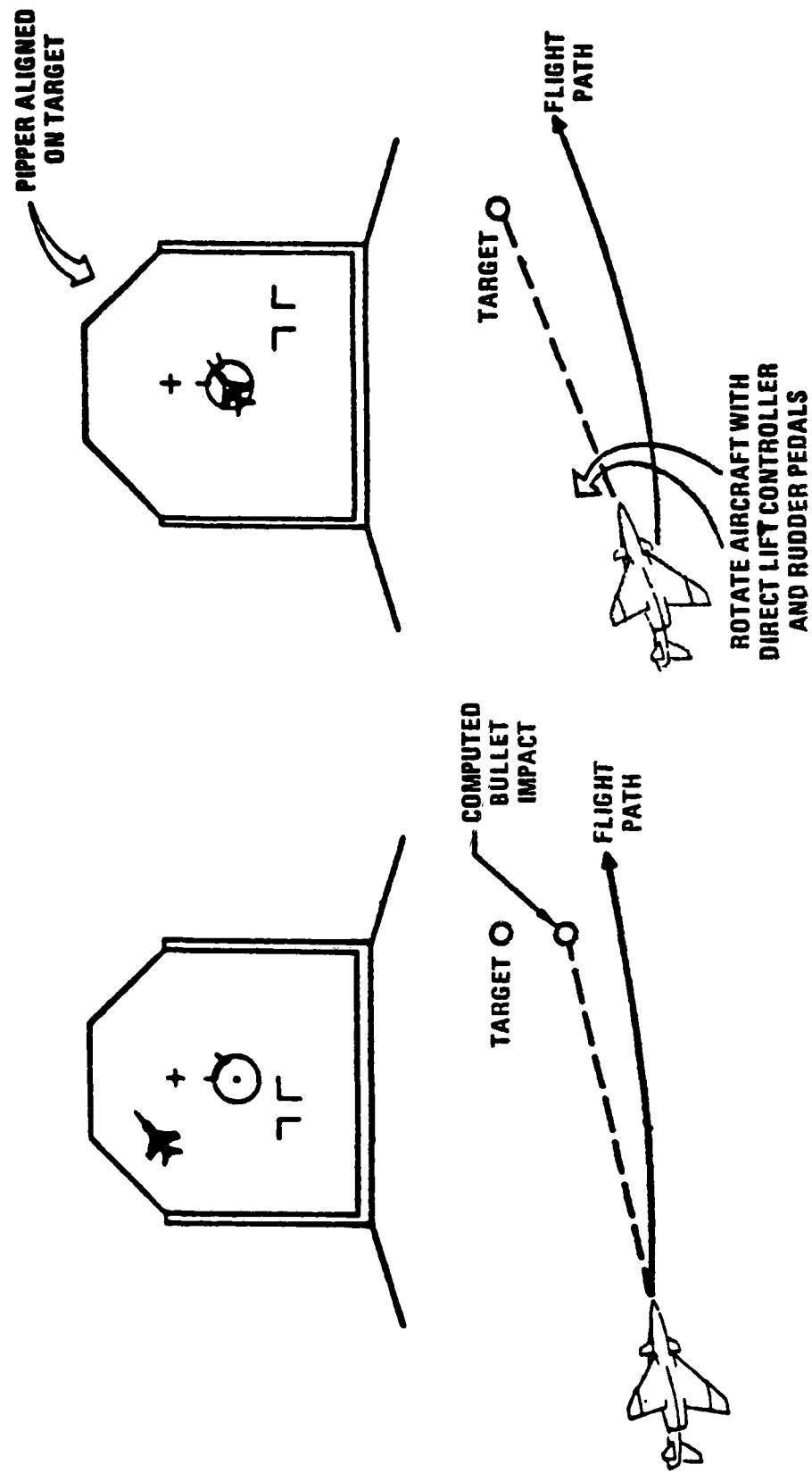


Figure 74. Effect of Fuselage Aiming On Air-to-Air Combat

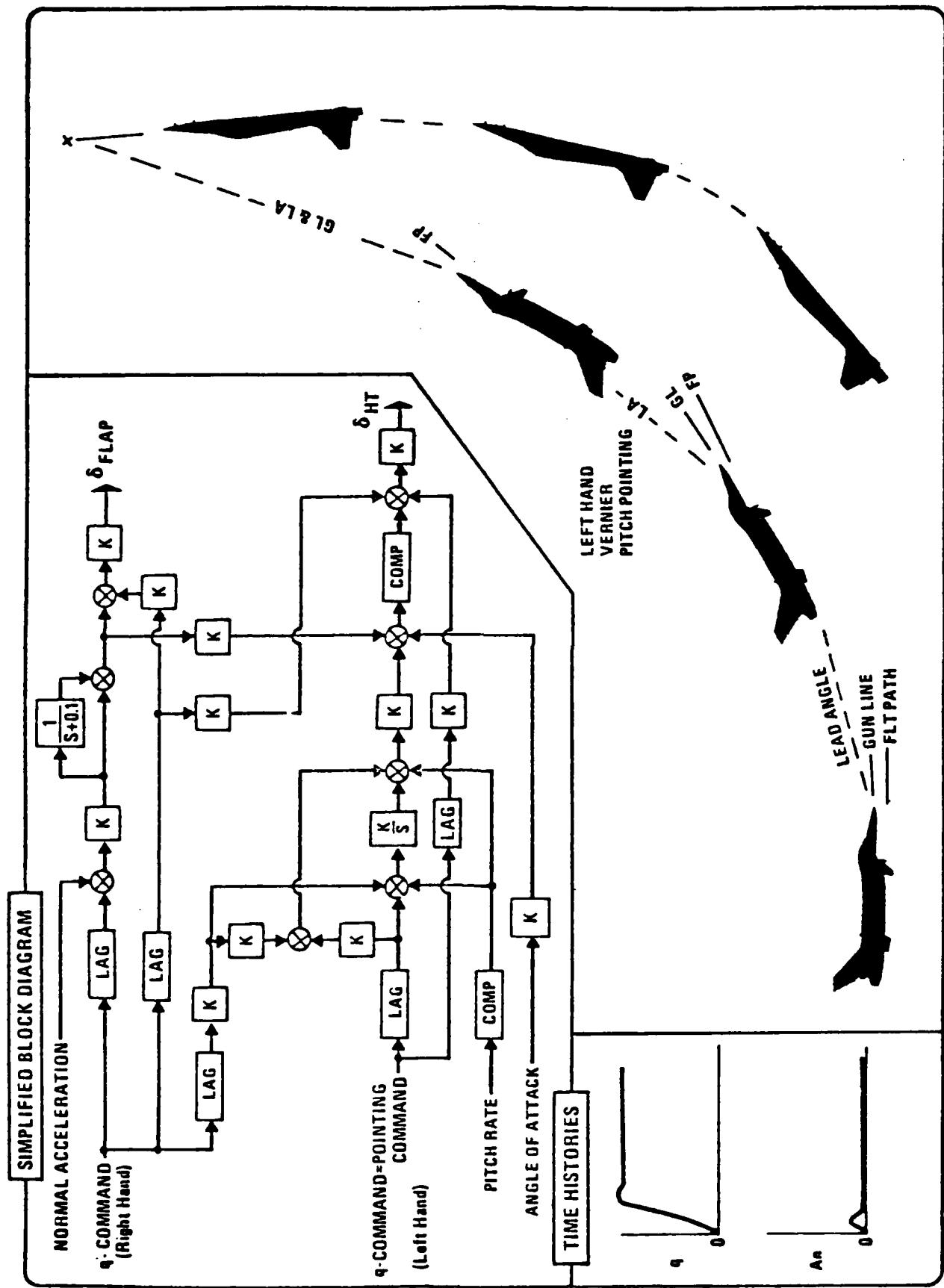


Figure 75. Air Combat Pitch Pointing Mode.

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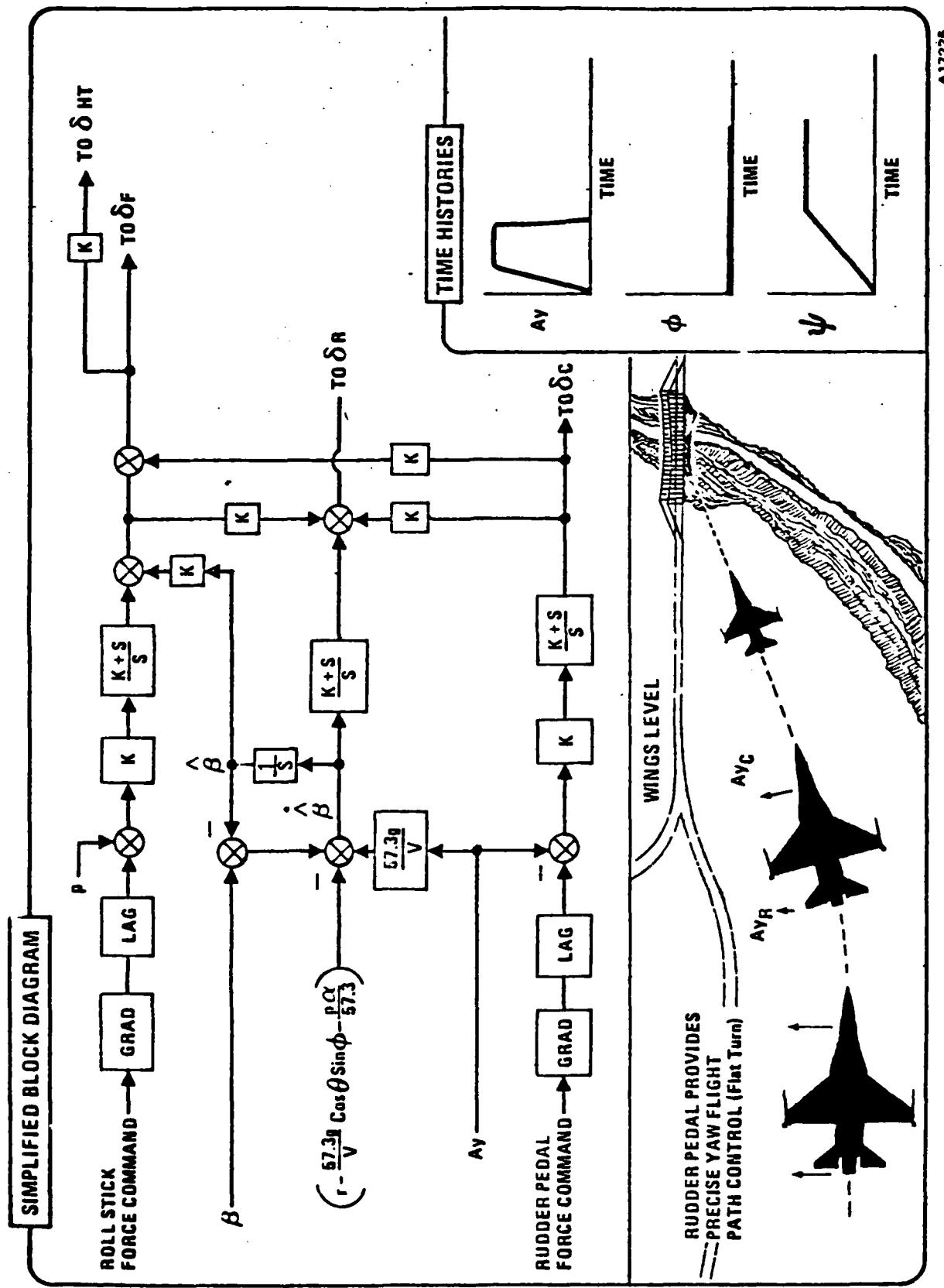


Figure 76. Air-to-Surface Flat Turn Bombing Mode.

The most important differences between the Standard Normal and Decoupled systems for lateral-directional augmentation is the choice of feedback states used to control yawing moments. The decoupled system used a synthesized beta-dot feedback for the following reasons: (a) Beta-dot is a key quantity that must be precisely regulated to maintain decoupling purity, and its direct use as a feedback will maximize the degree of control over it. (b) Better dutch-roll damping can be obtained from a beta-dot system in spite of the special decoupling interconnects that are required and tend to destabilize the system. This consideration is much more important because directional motions will be intentionally induced as a control mechanism during precision tracking tasks. (c) Better turn coordination can be obtained from the beta-dot system. (d) The beta-dot system provides a natural inner loop for an IFFC system. A maximum of 2 g's can be commanded in the direct sideforce maneuver. However, at most flight conditions the canard effectiveness or structural limits are not adequate to generate that amount of force while sideslip is being constrained to zero. Consequently, when the canards do saturate and the recoupling option has been selected, the canard-to-rudder decoupling signal is reduced to allow sideslip to develop. In most cases, this sideslip will generate enough additional sideforce to satisfy the pilot command.

The descriptions of the advanced flight control modes of the AFTI/F-16 were obtained from Reference 11.

THE ADVANCED FLIGHT CONTROL ACTUATION SYSTEM (AFCAS) PROGRAM

The development of Advanced Flight Control Actuation Systems (AFCAS) for next generation aircraft has been a joint undertaking by the Navy and the Rockwell International Corporation since 1972. The sixth and final phase of the program was completed in November 1979.

The complexity of flight control systems has increased until present initial costs and required maintenance time are approaching prohibitive levels. This situation is due primarily to the design philosophy that improvements and refinements are best achieved by adding on accessories and/or components to proven, traditional systems. Broad new approaches and technologies involving advances in power generation, transmission, control, and actuation will be required to alleviate complexity in future Navy aircraft. The AFCAS Program is a significant step in this direction.

Phase I of the AFCAS Program established that a direct-drive flow control valve, modular configured actuator, and a localized power package could be readily integrated into a computer-operated, fly-by-wire system. Adoption of AFCAS concepts should enhance flight control system maintainability, reliability, combat survivability, and lower initial costs, Reference 12.

Efforts to confirm the practicality of Phase I concepts were begun in Phase II with the design and fabrication of an engineering model, 8000 psi (55 MPa), control-by-wire, modular configured aircraft-type hydraulic servo actuator, Reference 13. Electrical inputs were applied to force (torque) motors employing cobalt samarium permanent magnets. Motor output was connected directly to single stage spool/sleeve type flow control valves. The force motors and flow control valves could be integrated into dual tandem, dual parallel, or single actuator configurations.

Phase III involved conducting laboratory performance tests on the engineering model actuator(s) built in Phase II, Reference 14. Static and dynamic tests were conducted on the force motors, motor/valve subassemblies, electronic drive unit, and actuator assemblies including dual system tandem, dual system parallel, and single system configurations. The dual tandem actuator was tested under load. Major achievements accomplished in Phase III were:

- Successful operation of a direct electrical control "muscle" actuator for primary flight control surfaces.
- Use of building-block elements to assemble dual tandem, dual parallel, and single actuator configurations.
- Successful operation of a control-by-wire hydraulic actuator, utilizing 8000 psi (55 MPa) operating pressure.
- Successful performance of a laboratory-type electronic drive unit which provided high immunity to circuitry failures.

In Phase IV, an 8000 psi (55 MPa) control-by-wire, modular rudder actuator was designed and fabricated for future flight testing on a T-2C airplane, Reference 15. Actuator design criteria were based on T-2C aerodynamic considerations, envelope constraints, and single system hydraulics. Actuator output was commanded by a single stage spool/sleeve valve driven directly by a permanent magnet force motor. The force motor was powered by an electronic drive unit which received inputs from a

force transducer in the rudder system and position transducers on the actuator. A localized hydraulic power unit supplied 8000 psi (55 MPa) pressure for the rudder actuator.

In Phase V, a direct-drive control-by-wire muscle actuator, powered by a localized 8000 psi (55 MPa) hydraulic system, was used to control the directional flight of a T-2C, Reference 16. Successful operation of the test installation represented a significant milestone in the development of advanced flight controls. No problems were encountered; the system functioned exceptionally well and pilot response was favorable. The test results confirmed analyses and laboratory investigations reported in References 12 through 15. The ease with which flight testing was accomplished verified that AFCAS-type systems can be designed, fabricated and maintained without special techniques or state-of-the-art advances.

In Phase VI the feasibility of Direct Digital Drive (D³) for an AFCAS was successfully flight tested in the T-2C aircraft, Reference 16. The test installation contained a digitally controlled direct drive rudder actuator, microcomputer, electronic drive unit, force transducers, and a localized 8000 psi (55 MPa) hydraulic power supply. The system met all laboratory and flight test objectives and demonstrated direct microcomputer control of primary flight control surfaces. Phase VI demonstrated an approach that will improve performance and reliability of FBW control systems by reducing system complexity.

DESIGN PHILOSOPHY OF THE AFCAS CONCEPT

Flight control systems need to be as simple, direct and foolproof as possible with regard to design, operation, inspection and maintenance. This requirement, as defined by MIL-F-18372, continues to be an excellent basis for the design philosophy of primary controls and needs to be stressed in evolving fly-by-wire (FBW) systems. Basically there are two ways of achieving the level of reliability required for fully powered primary flight controls. The first is to develop highly reliable parts for use in the systems. The second is to design reliable systems utilizing redundancy techniques. Neither approach leads to an acceptable solution of the immediate problem. Development and verification of components having the necessary reliability is a long term project while indiscriminate system redundancy leads to impractical complexity and unwieldy packages. A judicious blend of the two approaches is necessary to effect an acceptable near-term system design.

The first step of fundamental importance in achieving high reliability in FBW control systems is minimizing the number of component parts. Pursuing this approach leads to a simplified FBW system utilizing dualized hydraulic power systems. To achieve reliability equal to or exceeding that of its dual mechanical counterpart, the approach should follow the design philosophies used in conventional mechanical systems. They are:

- Failures are never abrupt but are progressive over periods of extended use.
- System failures do not result in hardovers.
- The existence of failures does not degrade total system operation.
- Design margins are in excess of normally used values.
- Design to minimize maintenance actions precludes "Murphy Failures" or human errors.
- Success of the system is attributed to subsystem elements with high designed-in reliability obtained with redundant configurations.

Fundamentally, design of FBW electronic circuits should center about the foregoing philosophy using feedback and redundant techniques. Present availability of small low cost electronic devices,

integrated circuits, (IC or LIC) permits consideration of this approach. Each function circuit would contain several paths for independent operation similar to the present cable system. The goal is a FBW system where the time between required maintenance actions approaches the aircraft overhaul period.

DESCRIPTION OF THE AFCAS CONCEPT

The AFCAS concept employs control of the primary surface actuators directly by an onboard digital computer or by a standby processor. Command signals are applied directly to the single stage control valve. Direct control of the surface actuator eliminates augmentation actuators. Actuator power is provided by 8,000 psi localized electrically or mechanically driven hydraulic packages in lieu of a centralized hydraulic system. Flexible building block components provides single or dual system surface actuators having the desired simplicity and hardware commonality for standard actuator classification. A direct drive single stage servo valve is utilized which eliminates contaminant sensitive small nozzles, flappers and jet-pipes. The aim is to provide simple, rugged, reliable components which can be integrated with an advanced digital flight computer into a complete FBW system for future advanced aircraft.

The control valve employs a highly reliable single stage design combining high performance with simplicity and ruggedness. A single moving part consisting of a spring centered, 4-way spool is driven by a high output torque motor. The basic valve is compatible with single and dual actuators. Replacement of the conventional two-stage hydraulic servo valve (8 ma control current) with a high force single-stage concept results in a significant improvement in reliability because small moving parts that are susceptible to minute fluid contamination are eliminated, and the fragile fine wire wound torque motor is replaced with a large ruggedized magnetic coil.

The evolution of solid-stage power amplifiers has resulted in devices, such as silicon controlled rectifiers and switches, that can now effectively and reliably handle the power required in going from low power control signals to the relatively high power level necessary for single stage valves.

This approach is centered on the elimination of the hardover potential by:

- Developing sufficient force on the valve spool to insure shear-out of all contaminants.
- Design of the valve torque motor coils such that an open or short will not cause a hardover.
- Design of torque motor drive circuitry which precludes hardover conditions due to component failures.
- Redundant feedback connections to preclude hardovers due to an open transducer.

The AFCAS configuration consists of a proportional control loop utilizing a 4-way spool and sleeve valve driven by a high output torque motor. The spool has two lands and is flow force compensated. Spool travel is 0.010 inches to insure null tracking over the entire operating temperature range. Flow forces are below one pound. The torque motor produces at least 40 pounds drive force on the spool at null position. The torque motor is kept dry to eliminate accumulation of contamination at the permanent magnet. The spool centering springs have a sufficiently high rate to insure positive centering at electrical zero and to provide sufficiently high frequency response. The valve housing is compatible with modular dual actuators utilizing rip-stop design principles. The dual spools have provision for rigid mechanical synchronization. The entire valve, including housing is made of steel. The valve assembly weighs less than 5 pounds. Figure 77 shows the salient features of the direct drive control valve package.

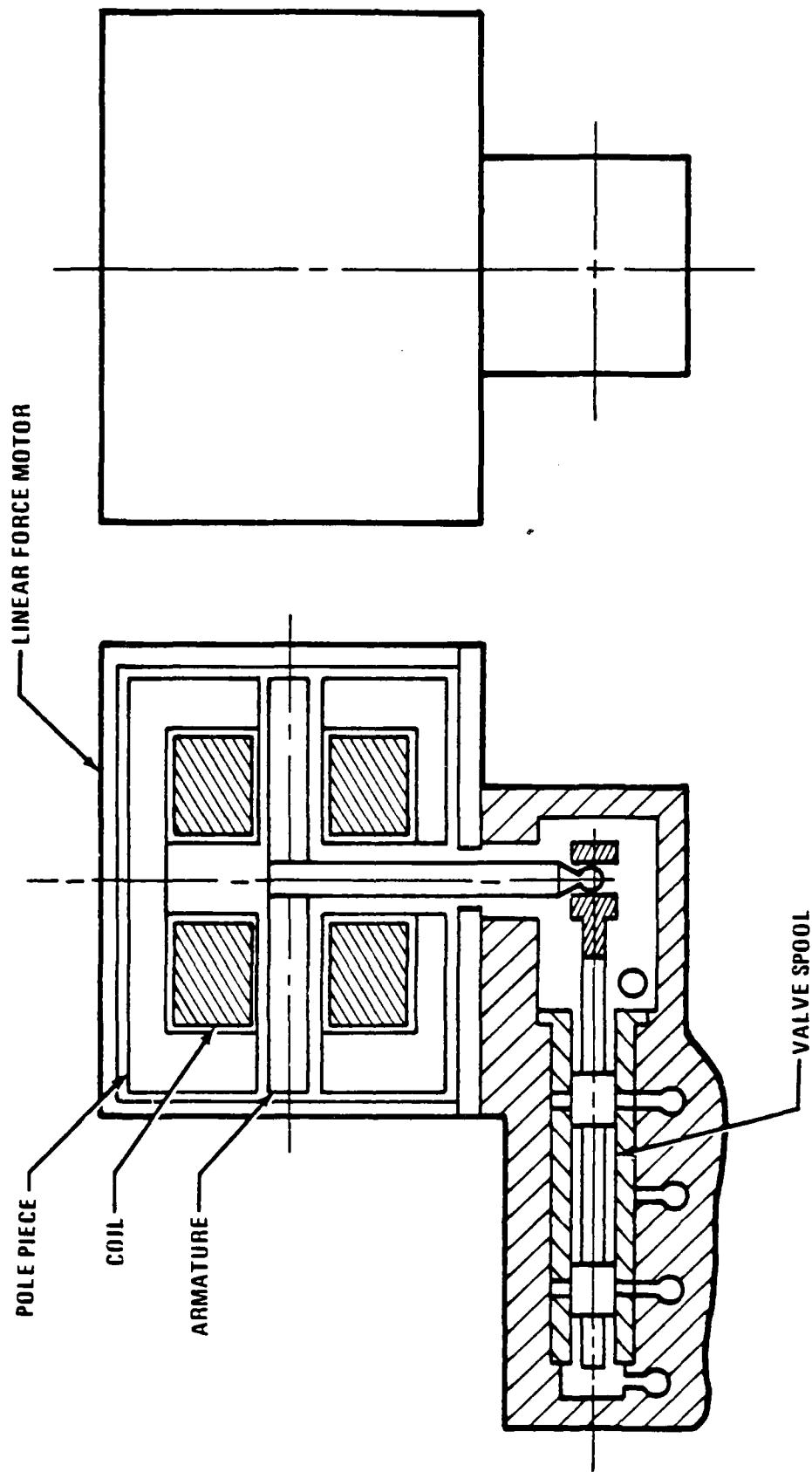


Figure 77. Direct Drive Control Valve Package

SERVO VALVE DESIGN REQUIREMENTS

The following general servo valve requirements were established during Phase I of the program to facilitate evaluation of the various options available and to provide a common basis for comparison. The requirements were kept as general and loose as possible to prevent unnecessary constraints.

- Single Stage

The valves shall incorporate only one hydraulic stage. This is a basic goal of the proposed approach. The aim is to eliminate small orifices and passageways which can clog and produce hard over failures.

- 8,000 PSI

The system operating pressure shall be 8,000 psi. VHP study effort to date indicates future aircraft should utilize high pressures to effect considerable savings in weight and space of hydraulic system components.

- Dual Tandem

The valve design must be compatible with dual tandem actuators. The proposed actuation system approach consists of two channels, and two hydraulic power output elements controlled by two electrical commands driving two valves connected in dual tandem.

- Shock

The valve shall be inherently rugged and able to withstand 1,000 g's shock in any axis.

- Weight

The weight goal for the entire valve assembly shall be less than 5 pounds. This represents a reduction of about 1 pound when compared to typical mechanical input surface actuator valves.

- Flow

The basic design shall provide flow rates up to 15 gpm by resizing orifice area gradients and/or flow compensation. The goal is to standardize valve parts. All classes of valves will be assembled using common housings, torque motors, mountings, etc. Adjustable mechanical stops shall be utilized to reduce the number of spool-sleeve designs to a minimum.

- Null Leakage

Null leakage shall be less than 0.5% of rated flow at 8,000 psi. This level is selected as a compromise between power loss, overlap, and manufacturing tolerances.

- Natural Frequency

The natural frequency shall be at least 35 Hertz. This value places the valve at least one order of magnitude beyond the actuator loop frequency thereby assuring minimal influence and ease of loop design.

- Hysteresis

The hysteresis shall not exceed 4 percent of the rated input signal.

- Dead Zone

Dead zone in the flow vs. input current characteristic curve shall be less than 4%.

- Valve Stroke

The spool stroke shall be 0.040 inch.

- Life

The valve shall withstand 4,500,000 cycles at rated pressure and 2% of rated flow, and 500,000 cycles at 100% rated flow.

- Spool Drive Force

The electromechanical transducer shall provide at least 40 pounds spool drive force at null. This requirement is a compromise between sufficient force to shear out material or contamination which could enter an orifice and reasonable size transducers.

- Linearity

The flow vs. current plot shall be linear within 15% of rated input current.

- Null Shift

The change in the null point shall not vary more than 5% throughout the operating environment.

AFCAS CONFIGURATIONS TESTED

Three types of actuation configurations were fabricated, assembled and tested during the third phase of the AFCAS program (reference 14) and are listed as follows:

- (1) Single System Actuator (Figure 78)
- (2) Dual System Parallel Actuator (Figure 79)
- (3) Dual System Tandem Actuator (Figure 80)

The "building block" concept developed in Reference 12 was used as a basis for the design of modular elements which can be assembled to form any of the three actuation configurations listed above. The two dual configurations are controlled by two mechanically synchronized Direct Drive Control Valve Assemblies. During the laboratory tests of the three actuator configurations, an electronic interface unit supplied the power required to drive the force motors. Independent current-drive circuits for each of the four windings in each motor were provided for redundancy. Figure 81 is a photograph of a force motor. Figure 82 shows the interface between the Electronic Interface Unit and a dualized Control-By-Wire Actuator.

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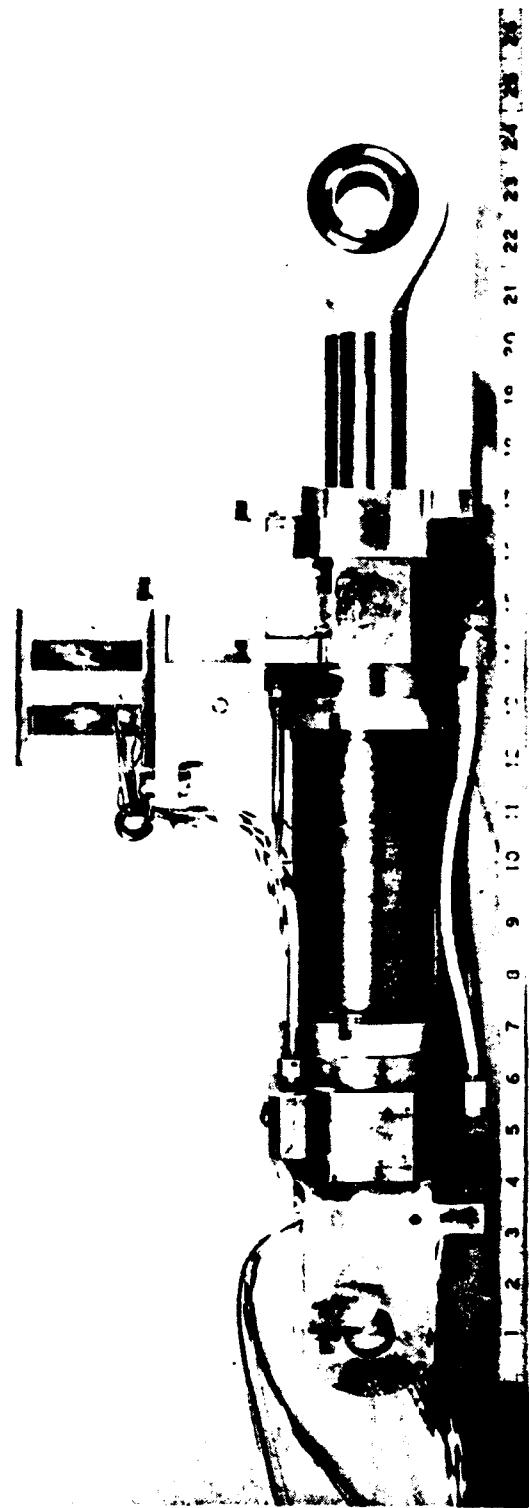


Figure 78. Single System Actuator

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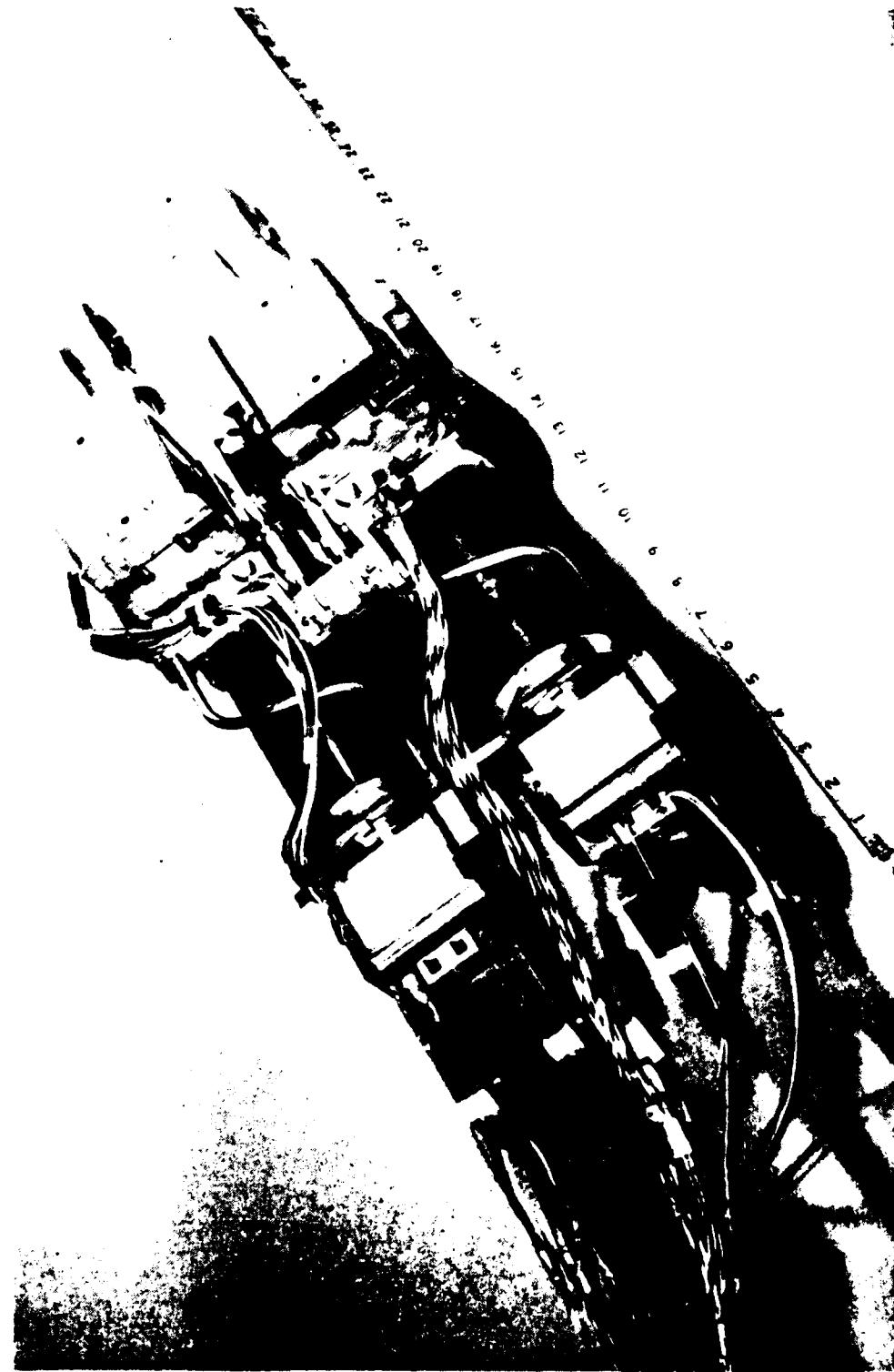


Figure 79. Dual System Parallel Actuator

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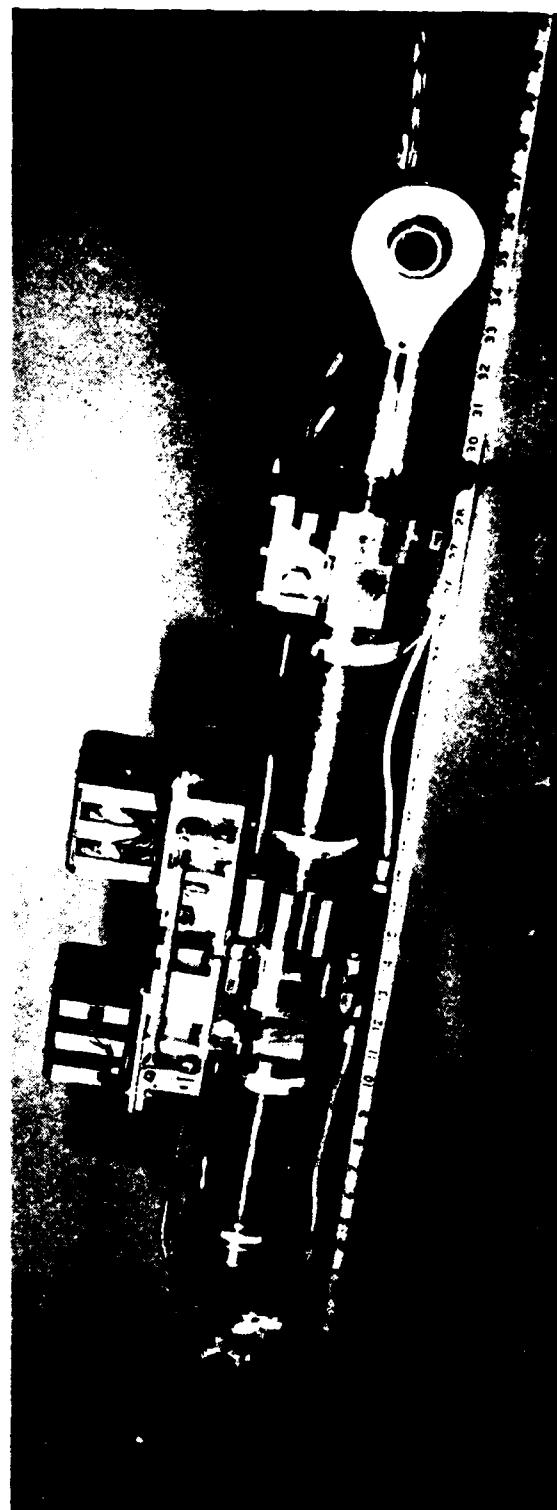


Figure 80. Dual System Tandem Actuator

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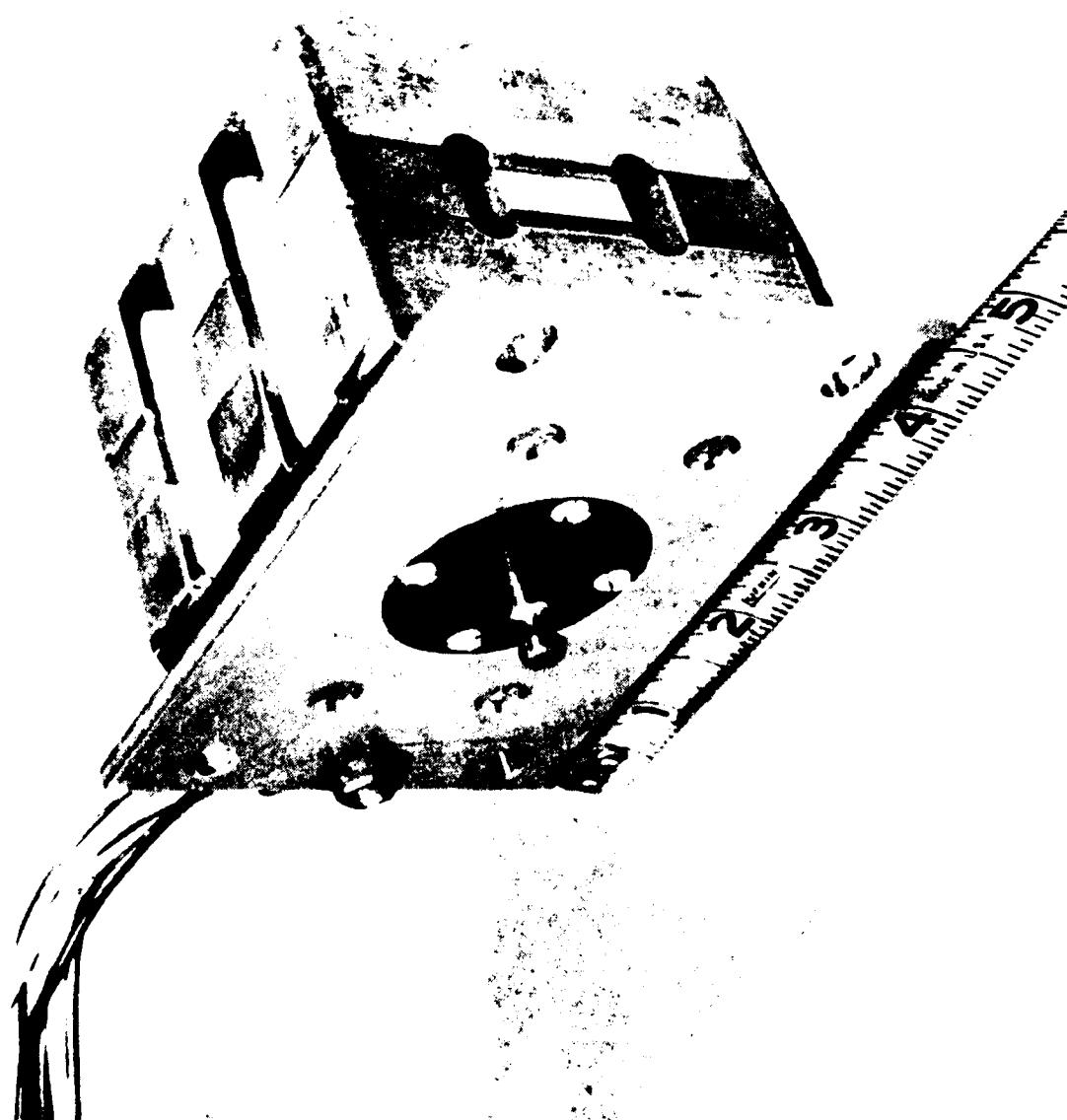


Figure 81. Force Motor

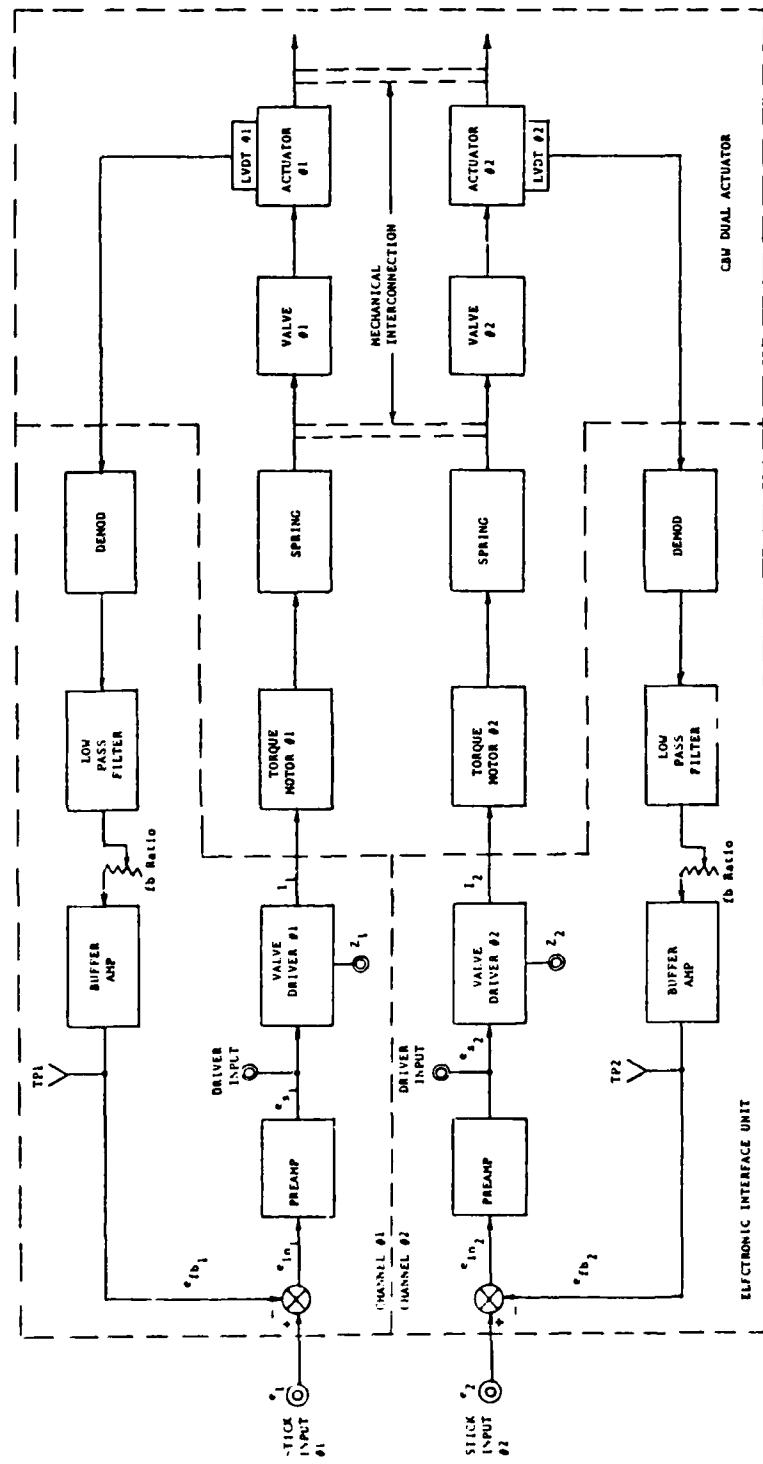


Figure 82. Block Diagram of Electronic Interface Unit and Dualized Control-by-Wire Actuator

The laboratory performance tests conducted during Phase III were successful and the major accomplishments achieved during this program phase are listed as follows:

- Successful operation of a direct electrical control muscle actuator for primary flight control surfaces.
- Use of building-block elements to assemble dual tandem, dual parallel, and single actuator configurations.
- Successful operation of a control-by-wire hydraulic actuator utilizing 8,000 psi (55 MPa) operating pressure.
- Successful performance of a laboratory-type electronic drive unit which provided high immunity to circuitry failures.

FLIGHT VERIFICATION OF THE AFCAS CONCEPT IN THE T-2C AIRCRAFT

In Phase IV, an 8,000 psi (55 MPa) control-by-wire, modular rudder actuator was designed and fabricated for future flight testing on a T-2C airplane, Reference 15. Actuator design criteria were based on T-2C aerodynamic considerations, envelope constraints, and single system hydraulics. Actuator output was commanded by a single stage spool/sleeve valve driven directly by a permanent magnet force motor. The force motor was to be powered by an electronic drive unit which received inputs from a force transducer in the rudder system and position transducers on the actuator. A localized hydraulic power unit was planned to supply 8,000 psi (55 MPa) pressure for the rudder actuator.

Figure 83 is a photograph of the Rudder Actuator Assembly designed and fabricated during Phase IV and used in the flight testing of the AFCAS concept during Phase V.

The objective of Phase V was to design, fabricate, and test a subsystem to verify the feasibility of the AFCAS concept in the flight environment. The test system was installed and flown in a T-2C twin engine turbojet trainer shown in Figure 84.

1. Technical Approach

The directional control system in a T-2C airplane was changed to a full-powered control-by-wire test installation containing:

- Hydraulic rudder actuator
- Electronic drive unit
- Localized hydraulic power unit
- Force transducer

The existing hydraulic system was altered to operate at two pressure levels: 3,000 psi (21 MPa) and 8,000 psi (55 MPa). Engine driven pumps powered the 3,000 psi system in the usual manner. A localized motor/pump unit was added to power the rudder system which was formerly operated manually by the pilot. The original 3,000 psi and newly added 8,000 psi systems shared the existing reservoir and return lines. The T-2C electrical system was altered to power the localized motor/pump unit and electronic drive unit. The modified system functioned the same as the basic T-2C system except the rudder was hydraulically powered instead of manually operated.

The original cable system between the rudder pedals and rudder was changed to incorporate the control-by-wire test installation. The rudder pedal cables were attached to a sector which was prevented from rotating by a force transducer. Force on the pedals was converted to a proportional

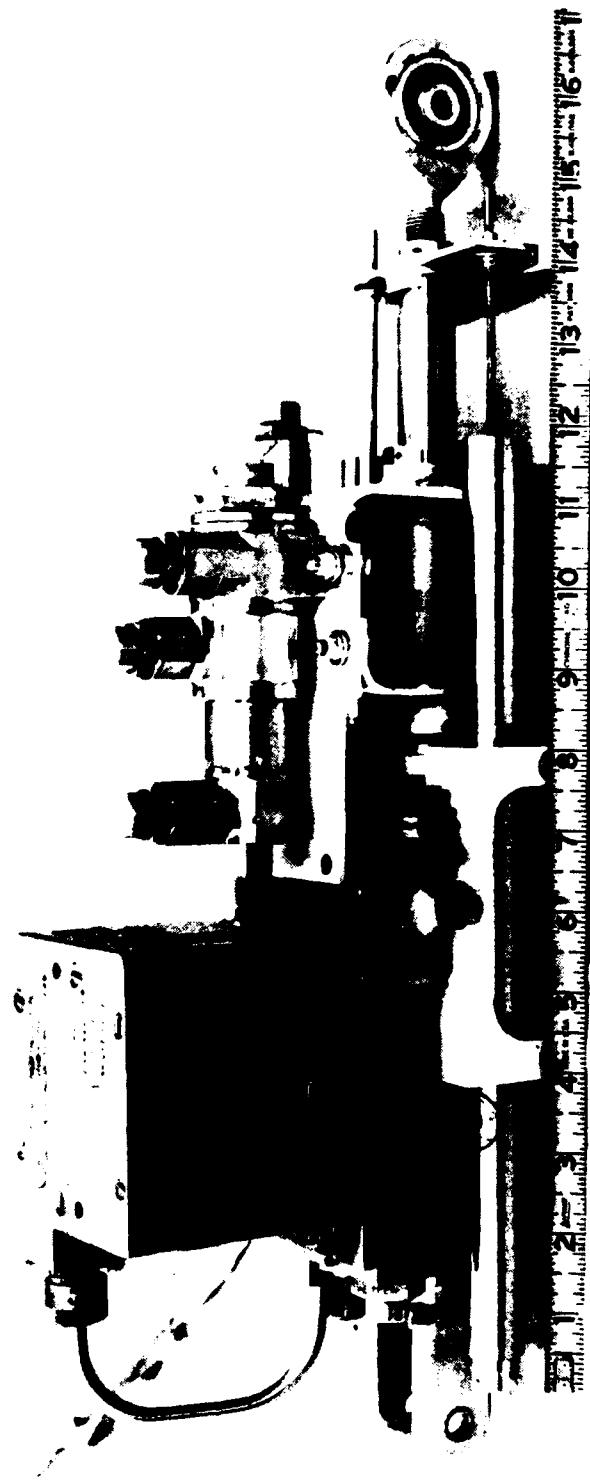


Figure 83. Rudder Actuator Assembly



Figure 84. T-2C Demonstrator Aircraft

electric voltage from the transducer. This command signal was conditioned by an electronic drive unit which powered a torque motor on the rudder actuator. The torque motor in turn operated a single stage flow control valve on the actuator.

The direct-drive, 8,000 psi rudder actuator designed and fabricated in Phase IV was modified to incorporate a bypass valve. This device allowed the rudder to seek the trail position if system pressure were lost. In the event of a "hard-over" type electronic failure, the pilot could permit the rudder to trail by turning the 8,000 psi motor/pump unit "off."

The electronic drive unit was designed, fabricated, and packaged to be a flightworthy assembly. The unit had dual channels with sub-circuits which were dualized. The circuitry was designed with redundancy features which provided high immunity to component failures.

Requirements were established for an 8,000 psi localized hydraulic power supply. The pump used was the same unit employed for flight testing in the Lightweight Hydraulic System (LHS) development program except delivery was reduced to match rudder actuator flow rates and to lower input power requirements. The pump was mated to an off-the-shelf, aircraft type 28 volt DC motor.

The force transducer incorporated in the test system was designed specifically for this application. The transducer utilized two linear variable differential transformers mounted in series.

All major components in the test installation were assembled in the laboratory for integration testing. Investigations were made to determine if detrimental pressure oscillations or surges were present. Motor current and system heat rejection were measured. Frequency response tests were conducted on the actuator/system. Nine hours of simulated flight testing were performed to evaluate the endurance capability of system components.

The test system was installed in a bailed T-2C with instrumentation for monitoring pressures, flows, temperatures, etc. Standard parameters such as air speed, altitude, engine RPM, etc., were also instrumented. Flight data were collected by photorecorder and telemetry systems.

Procedures were established for system checkout, ground demonstration, and flight testing. Approximately ten hours of flight time were logged on the test system at various altitudes and airspeeds. Pilot observations and instrumentation data were used as a basis for evaluating the AFCAS installation.

2. T-2C Airplane

The T-2C "Buckeye" is built by the Columbus Aircraft Division of Rockwell International Corporation. The Buckeye is a two-place, subsonic trainer powered by twin turbojet engines. The aircraft is designed for both land and carrier based operations. Distinguishing features include wide-track tricycle landing gear, straight tapered wings, and low slung intake ducts.

The T-2C is used as a basic trainer for military pilots, and is equipped for cross-country flight, night flying, and low altitude, high speed navigation exercises. Maximum level flight speed of the Buckeye is 465 knots (239 m/s) at 15,000 feet (4.6 km); the service ceiling is 45,000 feet (13.7 km). Take-off and landing speeds are in the range of 95 to 110 knots (49 to 57 m/s). A typical take-off gross weight is 13,000 pounds (5,900 kg).

Dual power sources are provided for the electrical, hydraulic, and air conditioning systems. The flight control system includes hydraulic full-powered ailerons, a boosted elevator, and an electric

trim system; rudder operation is manual. The aileron and elevator actuators are part of mechanical linkage connecting the pilot's stick to the control surfaces. Thus, in the event of a hydraulic system malfunction, control of the aircraft can be accomplished manually.

The T-2C has a 3,000 psi (21 MPa), Type II (-65 to +275°F) (-54 to +135°C) single hydraulic system. Two pumps, one on each engine, provide power to operate the landing gear, speed brakes, arresting hook, aileron actuator, and elevator boost package. The pumps are constant pressure, variable delivery, axial piston designs. Each pump is capable of delivering 4.9 gpm (18.5 L/m) at 7,800 rpm. Hydraulic fluid (MIL-H-5606) is supplied to the pumps by an air/oil type reservoir pressurized by engine bleed air. Fluid cleanliness is maintained by 5 micron absolute filters.

One pump can adequately handle all flow demands. However, if supply pressure should drop below 1,800 psi (12 MPa), a priority valve is used to insure operation of the aileron and elevator actuators. A cockpit controlled shutoff valve is installed in the aileron/elevator subsystem to permit simulating loss of power for training purposes. The landing gear and arresting hook can be lowered and locked by gravity, if desired. The wheel brakes have an independent hydraulic system.

Electrical power is supplied by two 28 volt DC 300 ampere starter-generators, one mounted on each engine. The generators are connected for parallel operation and power the primary bus. Output voltages are regulated for varying loads and engine speeds.

Two nickel-cadmium 24 volt re-chargeable batteries are used for engine starting and emergency DC power. The batteries are normally connected in parallel, but are used in series for engine starting.

A portion of the 28 volt DC power is converted to 115 volt 400 Hz AC power by two rotary inverters. Inverter No. 1 produces 500 volt-amperes for instruments; inverter No. 2 generates 1,500 volt-amperes for avionics and serves as a backup source for instrument power.

3. AFCAS Flight Test Installation

The directional (rudder) system in a bailed T-2C (BuNo. 152382) was changed from a manual to a full-powered control-by-wire system for the AFCAS program. Principal components in the test installation were:

- Hydraulic rudder actuator
- Electronic drive unit
- Localized hydraulic power unit
- Force transducer

Modifications required in the T-2C to accommodate the new installation are discussed in the following sections under four general headings: mechanical system, hydraulic system, electrical system, and instrumentation.

a. Mechanical System

Elements of the mechanical system are depicted schematically on Figure 85. The salient mechanical components of the T-2C rudder control system are listed as follows:

- Sector assembly
- Sector support

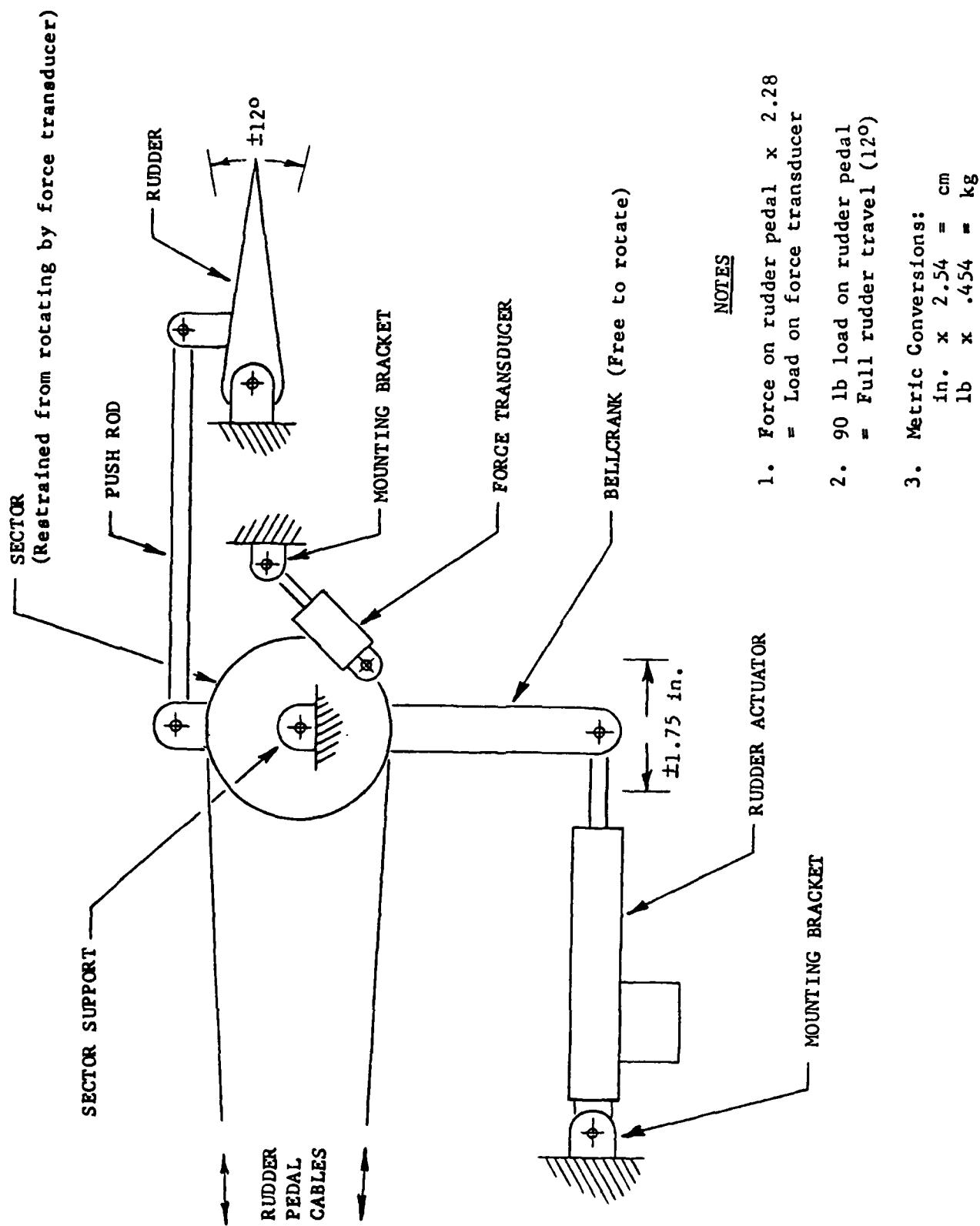


Figure 85. Schematic Diagram of Mechanical System

- Bellcrank assembly
- Rudder actuator mounting bracket
- Force transducer mounting bracket

The T-2C rudder has a travel of $\pm 25^\circ$. For safety reasons, rudder travel was reduced to $\pm 12^\circ$ in the test installation by limiting actuator stroke. This permits the pilot to land safely with a "hard-over" rudder, opposite engine out, and three knot cross-wind.

The relationship between pedal force and rudder travel is approximately 7.5 lb/deg (33.3 N/deg) of rudder movement or 90 lb (0.4 kN) for full travel (12°). Pedal displacement was small, approximately 0.50 in. (13mm), since the force transducer length changed only 0.025 in. (0.63mm) for full rudder travel (pedal displacement was due primarily to cable stretch). In the original manual control system, pedal displacement was approximately 4 in. (10.2 cm) for full rudder travel (25°).

The maximum hinge moment normally applied to the T-2C rudder is based on pilot strength and is 2,200 lb-in (249 N-m). Maximum rudder deflection a pilot can achieve thus depends on air loads present. The AFCAS rudder actuator can develop 13,000 lb-in (1,470 N-m). Because of the limited rudder deflection (12° max.) the high moment capability of the rudder actuator required only minor adjustment in the T-2C flight envelope to assure safety.

b. Hydraulic System

The original and modified hydraulic systems are compared schematically on Figure 86. Major changes required in the T-2C hydraulic system to accommodate the test installation were:

- Addition of an electric motor driven 8,000 psi (55 MPa) variable delivery pump
- Addition of an 8,000 psi control-by-wire rudder actuator and bypass valve
- Addition of a suction line from the reservoir to the 8,000 psi pump, pressure line from the pump to the rudder actuator, and actuator return line
- Addition of pump case drain return and shaft seal overboard lines
- Relief valve installed in the 8,000 psi system
- Heat exchanger installed in the 8,000 psi pump case drain line

The modified system is shown schematically on Figure 87. The 3,000 psi (21 MPa) and 8,000 psi (55 MPa) systems shared a common reservoir and common return lines. All major components, except for the rudder actuator, were located in the fuselage compartment above the engines.

c. Electrical System

The electronic drive unit was designed and fabricated by the Columbus Aircraft Division of Rockwell International Corporation. Circuit concepts employed in the unit were developed under company funded R&D projects. Innovative application of redundancy and feedback techniques permit EDU operation to be maintained with multiple component failures. Although the assembly was designed and fabricated to be suitable for flight, the EDU was nevertheless an experimental model. The assembly contained discrete components, test points, and external adjustments to facilitate data acquisition. This resulted in a much larger package than would be needed for a production unit. A production design EDU would have approximately 5% of the volume of the AFCAS unit.

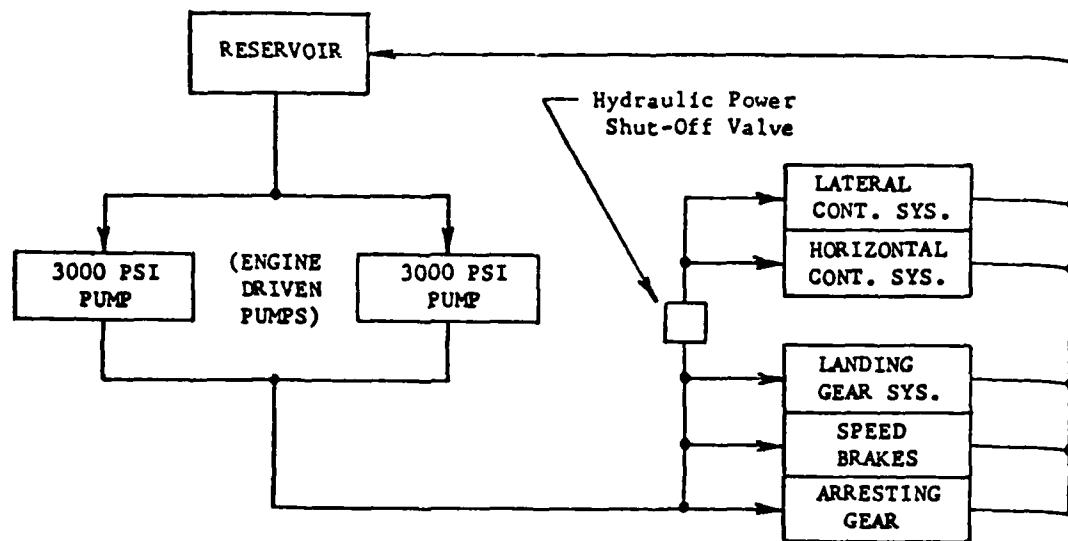
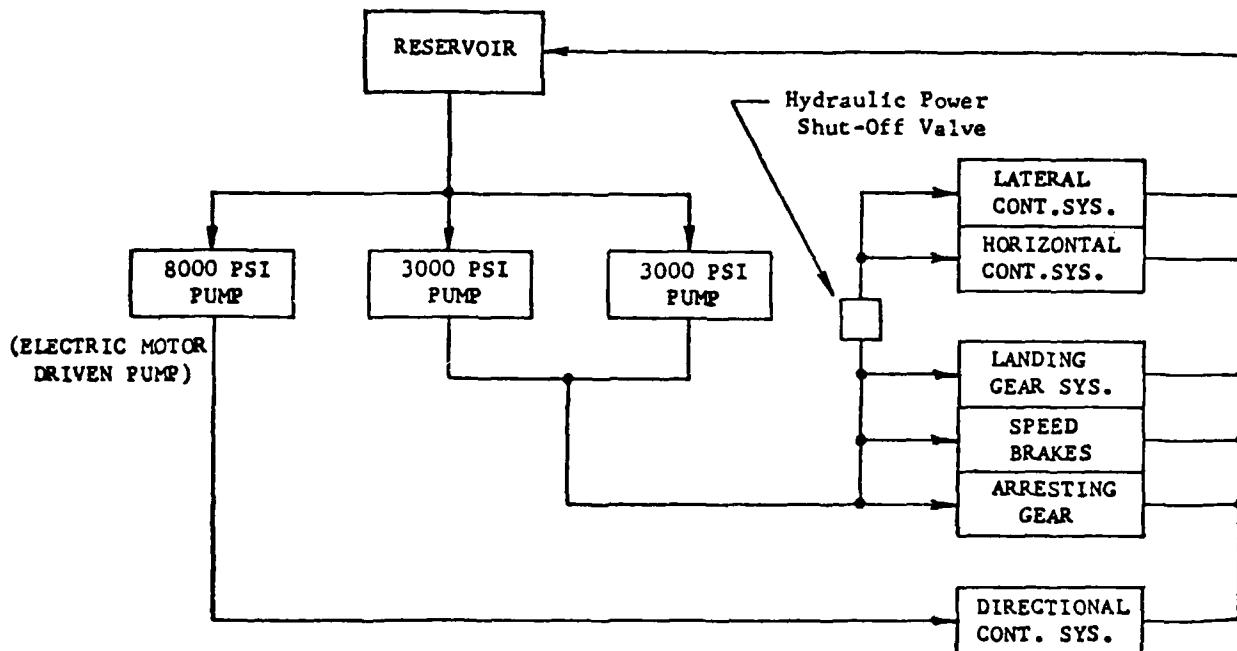
ORIGINAL 3000 PSI SYSTEMMODIFIED HYDRAULIC SYSTEM

Figure 86. Original and Modified Hydraulic Systems

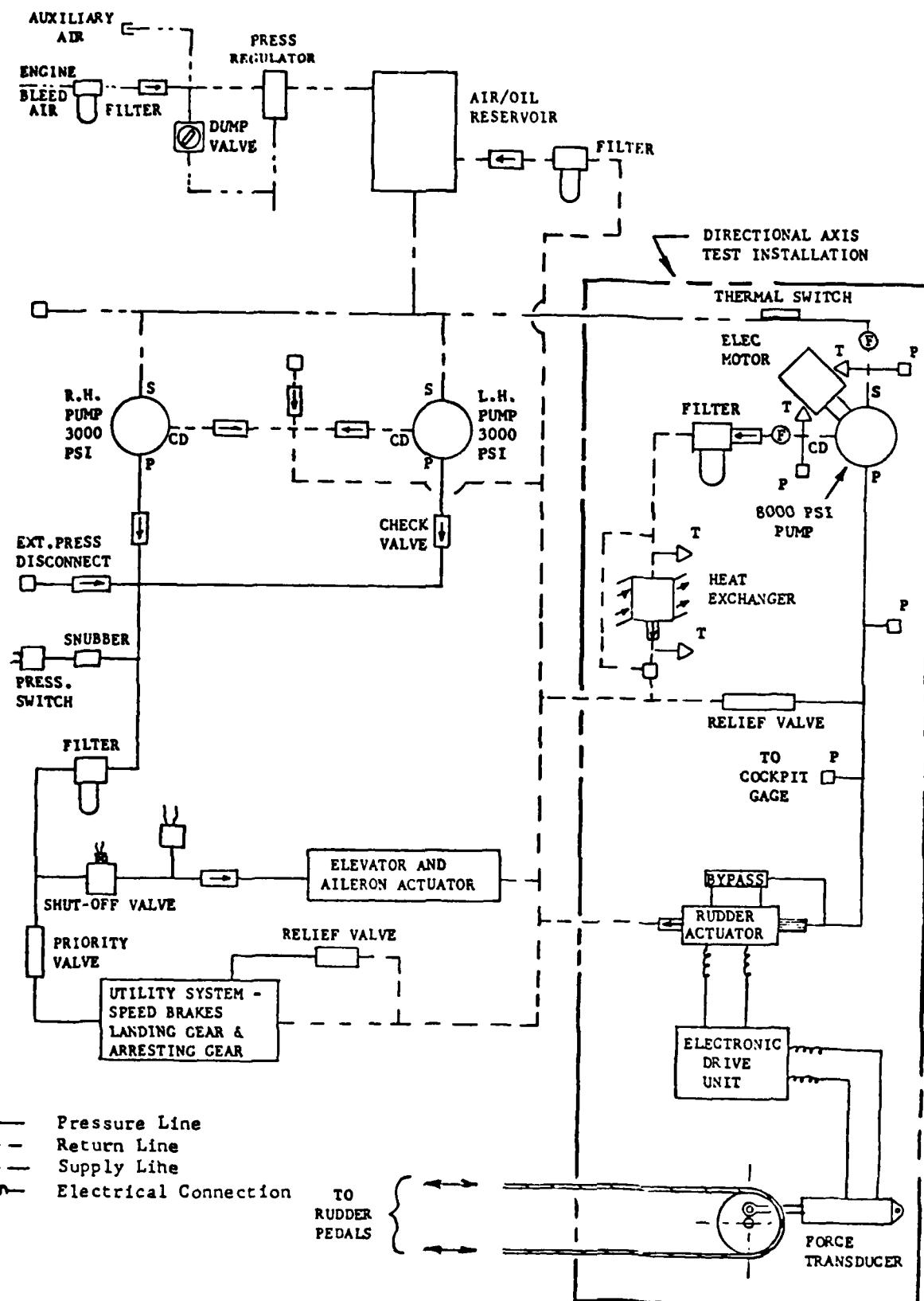


Figure 87. Schematic Diagram of Modified Hydraulic System

The EDU was powered by 115 volt 400 \sim AC and basically had two channels with dual sub-circuits (4 channels total). Bias pots were provided to adjust the input, feedback, and balance of each channel. Two power supplies provided \pm 15 VDC for the signal amplifiers and system transducers. All circuitry was contained on two identical printed circuit boards.

The force transducer housing assembly was designed and fabricated by CAD. The housing contains two DC-operated LVDT force transducers built by Schaevitz Engineering in Camden, New Jersey. The units have a maximum capacity of 500 lb (2.2 kN), a spring rate of approximately 8,000 lb/in (1.40 MN/m), and an output of 0.01 v/lb (0.002 v/N) in tension or compression.

A simplified block diagram of all elements in the system is shown on Figure 88. Pilot inputs are transmitted through the rudder pedals via cables, pulleys, and bellcranks to the force transducer located inside the vertical stabilizer. Gearing multiplies pilot effort by 2.28. Transducer output is the command signal (e_i) to the EDU. Amplifiers in the EDU process e_i with a feedback signal (e_{fb}) and power the force motor coils which drive the spool X_i in the control valve. The valve ports 8,000 psi hydraulic fluid to the rudder actuator in response to X_i . Actuator piston travel is sensed by position transducers having an output of 5 v/in; this is the feedback signal (e_{fh}). Actuator piston travel (\pm 1.75 in. max.) is converted through bellcrank and push rod to angular travel of the rudder (\pm 12° max.).

A simplified diagram of electrical components in the test installation is presented on Figure 89. System redundancy is illustrated on Figure 90. The system concept developed under CAD IR&D studies is flexible in that various levels of redundancy could be employed (as required) for other applications. AFCAS redundancy features include:

- Dual force (input) transducers
- Dual position (feedback) transducers
- Dual power supplies
- Quad electronics
- Feedback fault correction

Each of the four power amplifiers in the EDU employs current feedback with a highly reliable darlington power transistor configuration and independent power supplies. The circuitry is designed so that in the event an output stage fails "hard-over," voltage applied to a motor coil will not exceed its rated value. This limiting feature permits a subunit failure to be compensated or nullified by another subunit. Closed loop tests reported in Reference 14 verified that operation of the redundant subunits provided high immunity to component failures.

A math model of the idealized system is presented on Figure 91. The transfer functions are for "small signal" inputs and do not reflect fluid flow saturation limitations or motor current limitations imposed by coil inductance. System spring-mass effects (actuator loaded) were not included. Optimum loop gain was 90; this provided a theoretical band width of 14.3 Hz and a damping ratio of 0.7.

Performance characteristics of the test installation were higher than could be utilized in the T-2C rudder system. To assure satisfactory operation, AFCAS dynamics were matched with T-2C directional system dynamics. This was accomplished by lowering loop gain to 20 and adding high frequency roll-off filtering to reduce the possibility of system noise.

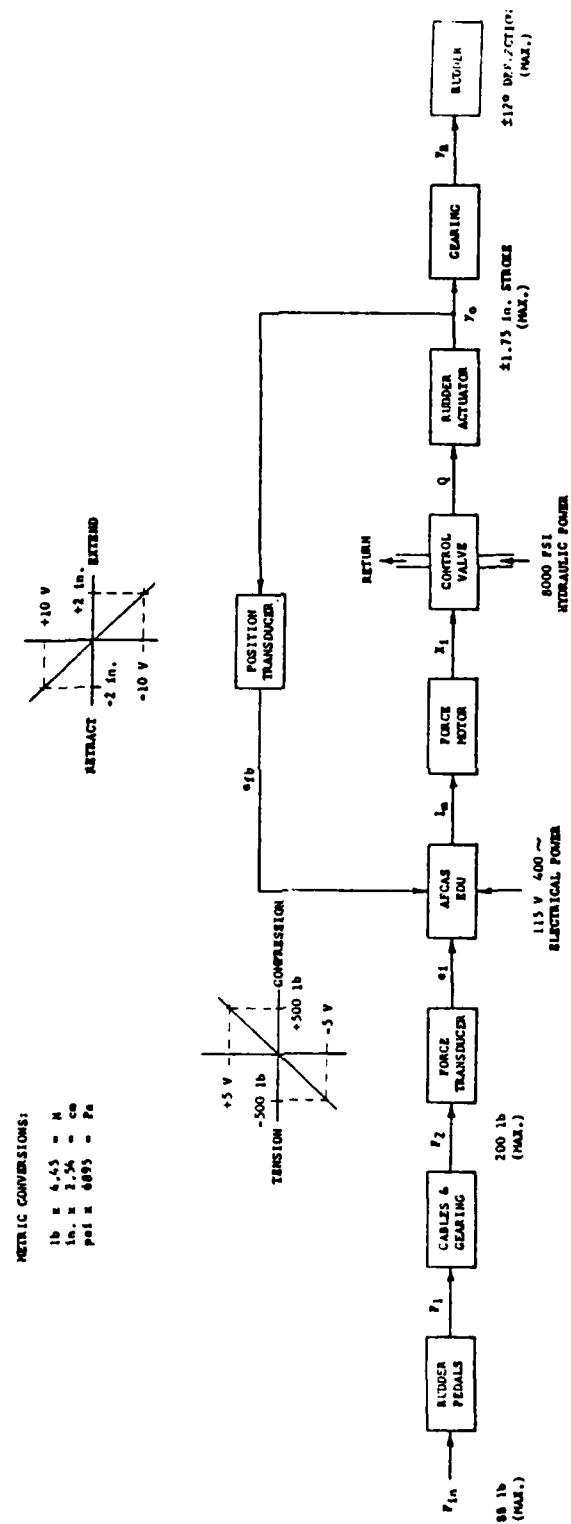


Figure 88. Block Diagram of System

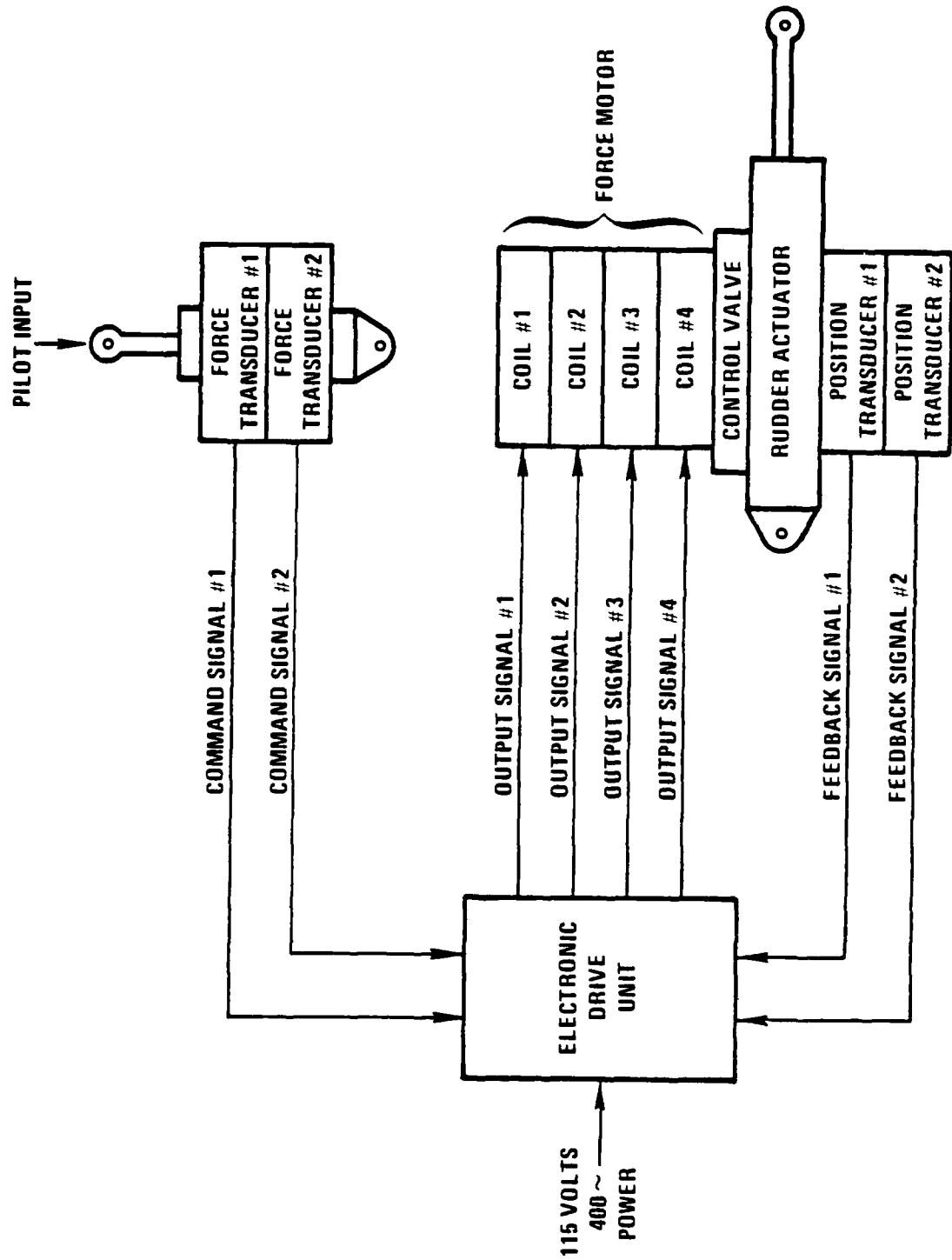


Figure 89. Simplified Diagram of Electrical Components

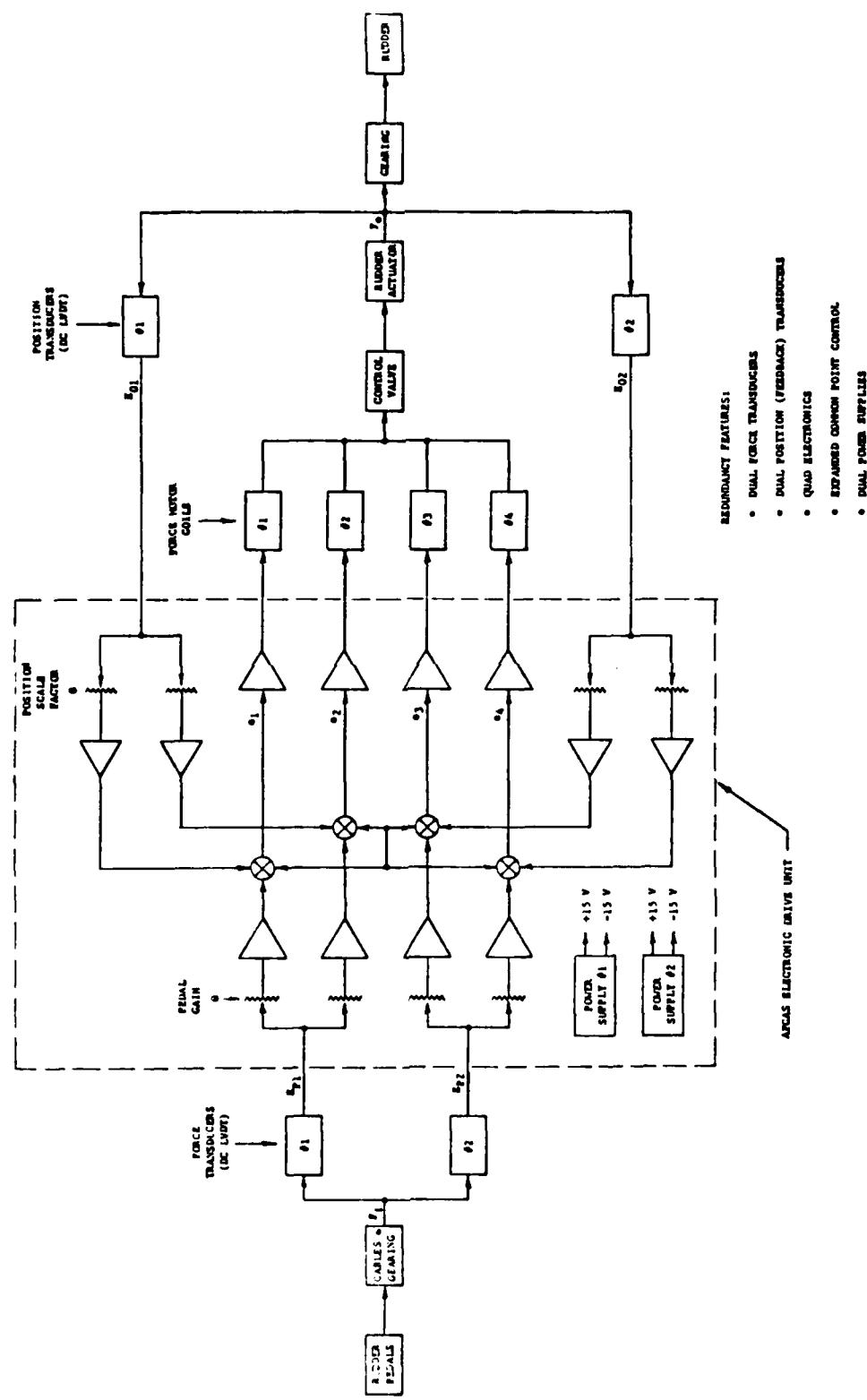


Figure 90. Simplified Diagram Showing System Redundancy

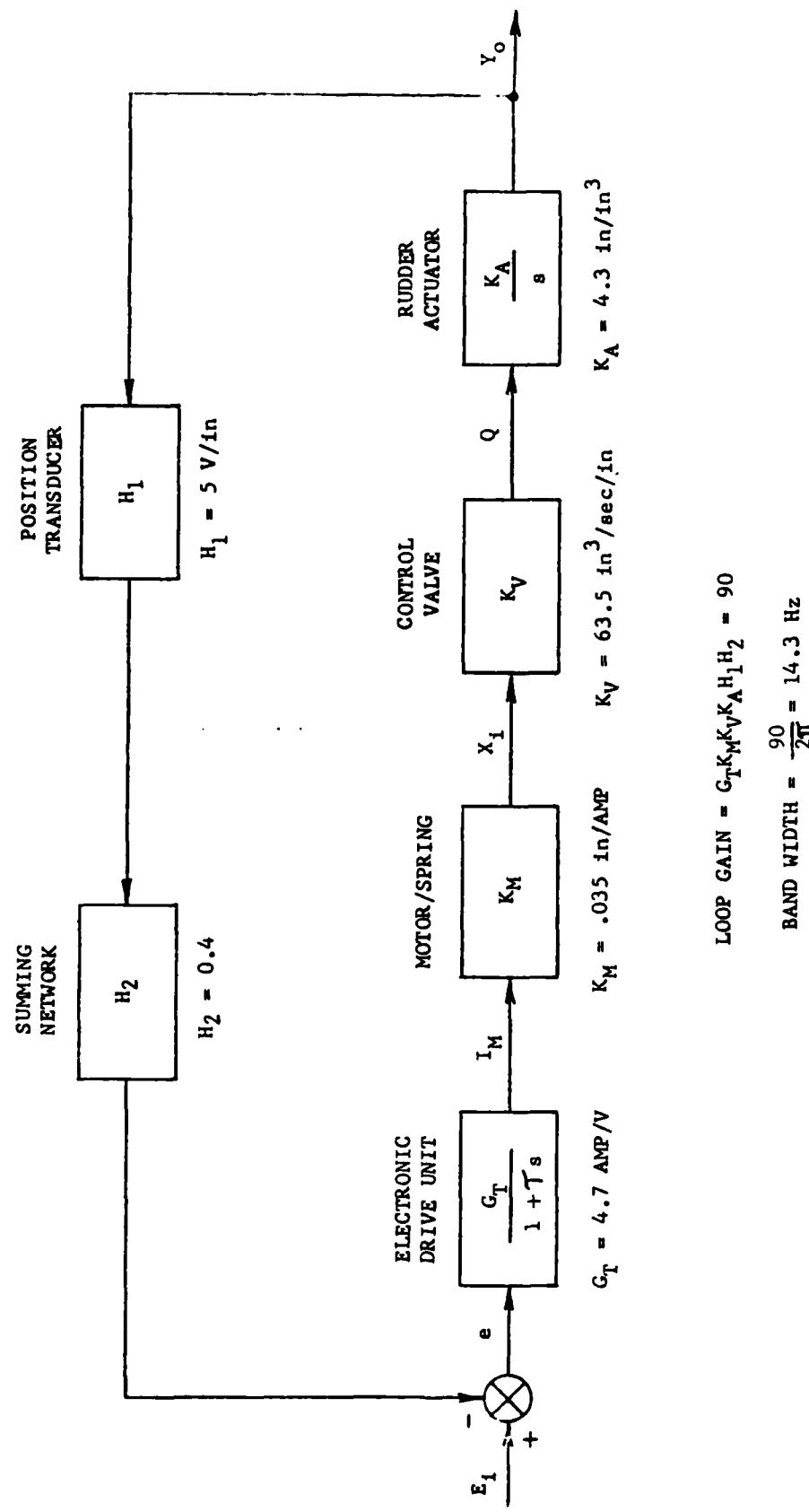


Figure 91. Mathematical Model of the AFCAS

d. Instrumentation

The T-2C was equipped with several flight data acquisition systems. Two were used in the AFCAS program: (1) an 18 channel telemetry system, and (2) a 21 hole photo recorder system. The telemetry oscillator package was located in the aft cockpit seat area; the photo recorder was installed inside the nose.

Telemetry data were recorded at the CAD Telemetry and Data Processing Center where a UHF receiving/tracking system provided real-time data acquisition and direct read-out on strip charts. Audio communication with the pilot was available for convenience and safety monitoring.

Pilot instrumentation controls were located above the cockpit instrument panel and on the control stick. Data in the two recording systems were related by means of correlator numbers printed on the photo recorder film, and correlator blips on the TM strip chart. A correlator counter could be read by the pilot for reference purposes.

New equipment installed to permit the pilot to monitor and control the AFCAS system were:

- An indicator was provided for direct readout of the motor/pump discharge pressure
- A switch was provided to turn the motor/pump unit "on" and "off"
- An "oil hot" light was set to illuminate when hydraulic fluid in the motor/pump suction line exceeded approximately +200°F (93°C)

4. Flight Plan

The primary objective was to verify the feasibility of the Advanced Flight Control Actuation System (AFCAS) concept by flight testing a control-by-wire, direct-drive actuation system powered by a localized 8,000 psi (55 MPa) motor/pump unit. Demonstration of flying qualities was not part of the program, however, pilot comments were encouraged. Ten flight hours were expected to be sufficient to evaluate AFCAS performance, confirm prior analyses and laboratory testing, and provide a measure of confidence in system reliability.

The flight plan was designed to determine directional control characteristics at several altitudes up to 30,000 ft. (9.1 km) and various speeds up to 340 knots (174 m/s). The first two flights were dedicated to confirming satisfactory operation. Subsequent flights were scheduled to evaluate system performance and reliability while accumulating 10 flight hours. Flight plan details are given in Appendix A of Reference 16.

5. Flight Test Results

The AFCAS flights are summarized on Table V of Reference 16. Two pilots participated in the program and prepared reports detailing each test flight. Additional comments were made during flight de-briefings. Both pilots stated that performance of the AFCAS test installation was completely satisfactory. Comments made by the pilots concerning their flights were:

- The AFCAS installation worked exactly as designed
- No malfunctions occurred
- System pressure was steady
- Hydraulic fluid temperatures were normal

- Directional control response was judged to be superior to the production T-2C
- Pilot adaptation to "force control" of the rudder was quickly and easily acquired. Reaction of the aircraft provided the clues to close the loop.
- The force system had an advantage during take-offs and landings in high cross-winds. The fixed pedals provide full rudder and allow much easier braking (in combination) without severe leg and foot extension that is required for conventional deflection controls.

6. Summary

A direct-drive control-by-wire muscle actuator, powered by a localized 8,000 psi hydraulic system, was used to control the flight of a T-2C. Successful operation of the test installation represented a significant milestone in the development of advanced flight controls. No problems whatsoever were encountered; the system functioned exceptionally well and pilot response was favorable. The test results confirmed analyses and laboratory investigations reported in References 12 and 15. The ease with which flight testing was accomplished verified that AFCAS type systems can be designed, fabricated, and maintained without special techniques or state-of-the-art advances.

The AFCAS concept is intended for application to automatic, computer operated flight control systems. The AFCAS flights described in Reference 16 did not demonstrate the full performance capabilities of the test hardware since the T-2C did not have computer operated controls. Company funded investigations at the Columbus Aircraft Division verified the feasibility of controlling AFCAS actuators directly by a digital computer.

FLIGHT VERIFICATION OF DIRECT DIGITAL DRIVE FOR AN AFCAS IN THE T-2C AIRCRAFT

The objective of the sixth and final phase of the AFCAS program was to demonstrate that AFCAS-type actuators can be directly controlled by a digital computer. The computer control was demonstrated in system laboratory tests and in flight tests in a T-2C twin-engine turbojet trainer.

1. Technical Approach

The directional control system of a T-2C aircraft was changed to incorporate a full-powered Digital Fly-By-Wire (DFBW) mode with an Analog Back-Up (ABU) mode. The test installation contained:

- Hydraulic rudder actuator
- Localized hydraulic power unit
- Digital microcomputer
- Electronic drive unit (EDU)
- Associated sensors, wiring and power supplies

The original cable system between the rudder pedals and rudder was changed to incorporate the fly-by-wire test installation. The rudder pedal cables were attached to a sector which was prevented from rotating by a force transducer. Force on the pedals was converted to a proportional electrical signal from the transducer. This command signal was supplied to a microcomputer where it was summed with a feedback signal, and processed into a pulse width modulated (PWM) error signal. The PWM signal was power amplified in the EDU which powered to torque motor of a direct drive hydraulic rudder actuator. The modified system provided a microcomputer controlled, hydraulically powered rudder, instead of the manually operated rudder of the basic T-2C aircraft.

The hydraulic system, the direct drive actuator, the EDU, the LVDT actuator position feedback transducer, and the pedal force transducers were installed and flight tested in the T-2C during

Phase V. The Phase VI system was designed so that the signals from the transducers could be switched from the microcomputer unit directly into the EDU to provide an ABU mode with the same control capability as the Phase V flight system. The ABU mode could be selected manually, or selected automatically if the microcomputer monitor detected abnormal operation.

The safety provisions of the Phase V program were included in this phase. The direct-drive 8,000 psi (55 MPa) rudder actuator, designed and fabricated in Phase IV, was equipped with a bypass valve. This device allowed the rudder to seek the trail position if system pressure were lost. In the event of a system failure, the pilot could permit the rudder to trail by turning the 8,000 psi (55 MPa) supply "off."

All major components in the test installation were assembled in the laboratory for integration testing. System operation was verified in the laboratory prior to aircraft installation. Frequency response tests and temperature-altitude tests were performed simulating aircraft operation.

The test system was installed in a T-2C with instrumentation for electrical and hydraulic operation. Standard parameters such as air speed, altitude, engine rpm, etc., also were instrumented. Flight data were collected by photorecorder and telemetry systems.

Procedures were established for system checkout, ground demonstration, and flight testing. More than four and one-half hours of flight time were logged on the test system at various altitudes and airspeeds. Pilot observations and instrumentation data were used as a basis for evaluating the test system.

2. Description of the AFCAS Direct Digital Drive (D³) Test Installation

The fly-by-wire rudder control system test installation, originally installed in the T-2C aircraft during Phase V of the AFCAS program was modified to test a digital microcomputer generated PWM valve drive signal (Phase VI of the AFCAS program). Principal components in the test installation are:

<ul style="list-style-type: none"> ● EDU ● Localized Hydraulic Power Unit (8,000 psi) ● Force Transducers ● LVDT Position Transducers 	Previously installed and tested per Phase V of the AFCAS program.
<ul style="list-style-type: none"> ● Microcomputer Assembly ● Microcomputer Power Supply 	Installed and tested per Phase VI of the AFCAS program.

Two modes of system operation are provided, the DFBW mode and the ABU mode. In the DFBW mode, the microcomputer converts the pedal force command and rudder position feedback outputs into digital signals which are summed, amplified, and converted into PWM signals. The PWM signals are sent to each of two channels in the EDU where the signals are amplified and power converted into four torque motor PWM currents. In the ABU mode, the pedal force commands and rudder position feedback outputs bypass the microcomputer and are connected directly to the EDU where they are summed, amplified, and power converted into four torque motor currents.

Figure 92 shows a simplified block diagram of the T-2C Direct Digital Drive AFCAS test installation. The DFBW engage/disengage functions are implemented by a cockpit control switch located adjacent

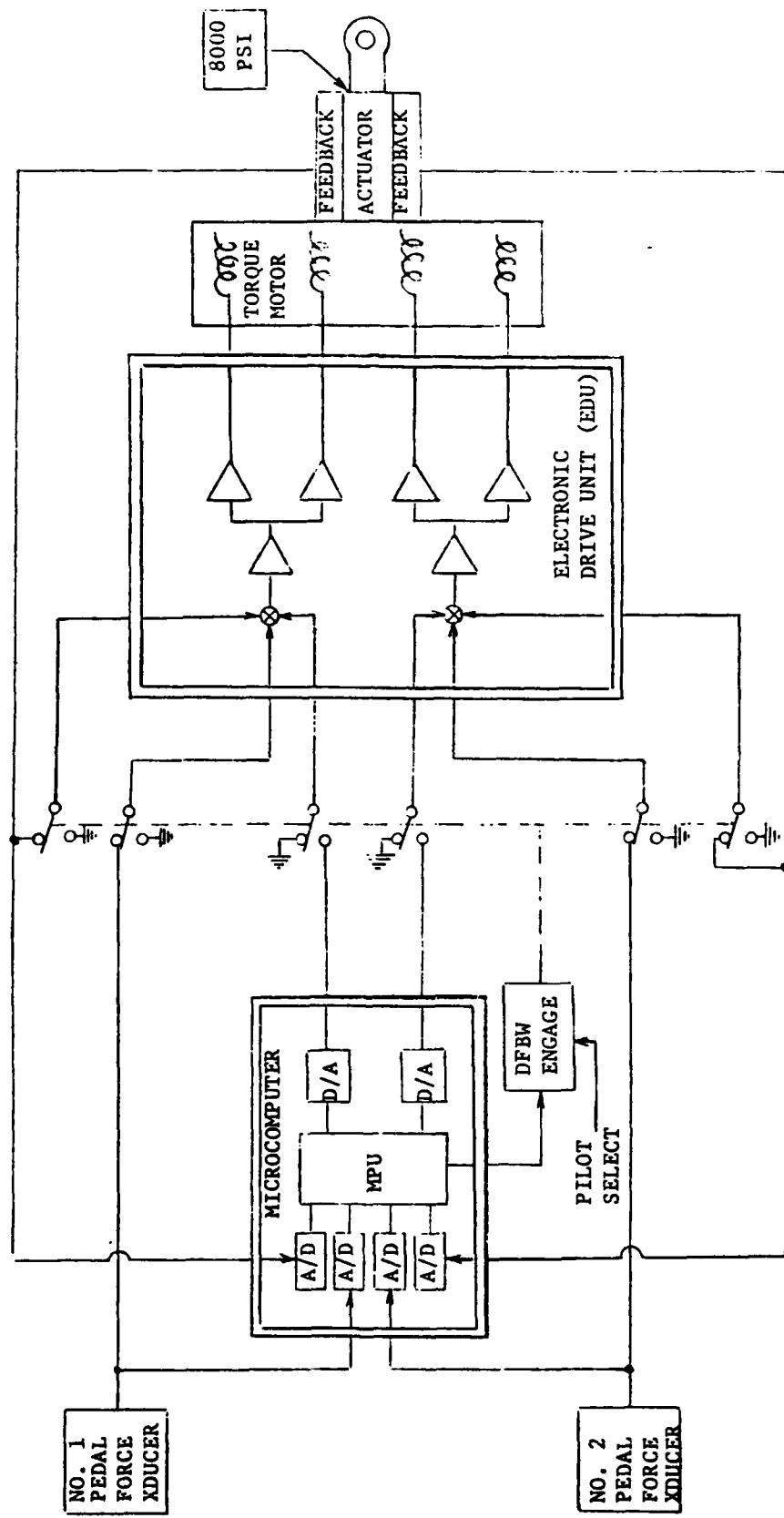


Figure 92. Simplified Block Diagram of T-2C AFCAS Direct Digital Drive (D3) Test Installation

to the AFCAS power switch on the Pilot's Auxiliary Instrument Control Panel. After the AFCAS analog system has been engaged, DFBW control will be selected by momentarily holding the DFBW engage switch to "ON". The DFBW control relays will energize to switch the transducer outputs to the microcomputer, and connect the microcomputer output to the EDU. When the microcomputer functions are operating correctly, a +28 volt DC power ground will be supplied to the holding coil of the cockpit DFBW engage switch. Disengagement of DFBW control will result from the following: manually selecting the DFBW engage switch to "OFF", automatically by the loss of the microcomputer supplied ground, or by manually selecting the AFCAS power switch to "OFF".

a. Microcomputer Assembly

The microcomputer assembly is housed in an enclosed unit, and consists of the following subassemblies:

- Motorola Mono-Board Microcomputer Module
- Analog-To-Digital (A/D) Converter Module
- Digital-To-Analog (D/A) Converter Module
- Card Cage & Mother Board Assembly
- Signal Conditioning Board

The mono-board microcomputer module is a complete computer-on-a-board having all the processing and control required for a microcomputer-based system. It incorporates the MC 6800 MPU, 1 K of Random Access Memory (RAM), provisions for 4 K of Programmable Read Only Memory (PROM), timing and control, buffers, an Asynchronous Interface Adapter (ACIA) and two Peripheral Interface Adapters (PIA).

The A/D converter module consists of eight channels of A/D conversion of which four are utilized. The D/A converter module consists of four channels of D/A conversion of which three are utilized.

The signal conditioning board contains four channels of sensor signal conditioning and a relay driver that interfaces the microcomputer monitor output with the system control logic.

Additional information on the microcomputer assembly is contained in Appendix A of Reference 17.

A separate power supply converts single-phase, 115 VAC, 400 HZ to + 5 VDC, and \pm 12 VDC to power the microcomputer assembly.

b. Software Description.

Software was developed to enable the microcomputer to perform two basic functions; a command/feedback control function and a control monitor function.

The command/feedback control function sums the pilot command and rudder position signals to produce an output signal proportional to the difference to drive the actuator.

The control monitor function measures the level of error between the pedal command and the rudder actuator position feedback, and if a preset level is exceeded for a given period of time, the engage command will be removed. Actuator control will then revert to the ABU mode. A continuous check is also made on the transducer input A/D conversion hardware by comparing the two digital feedback signals with each other and in a similar manner comparing the digital pedal signals. Any differences exceeding preset levels for a given period of time will result in switching system control to the ABU mode.

The control monitor function was incorporated as the Motorola microcomputer is a single channel device which could generate a "hardover" command under certain failure conditions. The dual-channel redundancy of the ABU mode prevents a "hardover" command of the rudder even if a pedal transducer or rudder position transducer fails in a "hardover" condition.

c. Program Modules

The DFBW Microcomputer Program Flow Chart, Figure 93, illustrates the modular nature of the software and the sequence in which the modules function. The program modules were designed, coded, and initially checked as individual entities prior to being integrated.

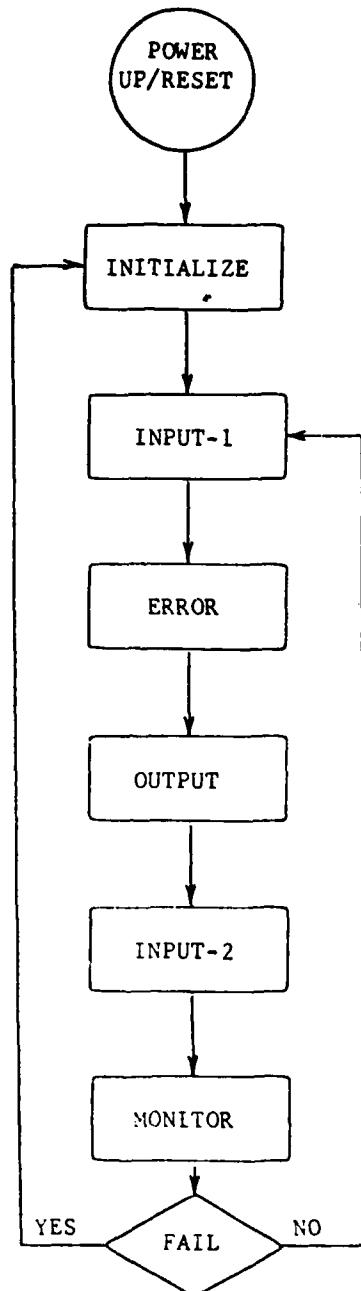


Figure 93. Digital Fly-By-Wire Microcomputer Program Flow Chart

Following is a brief description of the program modules:

Initialize - The Initialize module sets the D/A Converter (DAC) channel 4 to provide +5 VDC to hold in a relay. The relay, in turn, holds the DFBW Engage switch in the engage position. This module also sets timing counters to ensure that the Monitor function does not immediately turn off the DFBW Engage switch.

Input 1 - The Input-1 module, as the first in the repetitive loop, is used to start the PWM output signals. This is done by setting both DAC-1 and DAC-2 at +10 VDC. It then controls the A/D conversions of pilot command (CMD1) and rudder position (POS1). Inputs are scaled so that full scale, $\pm 12^\circ$ of rudder is ± 5 VDC, which is one-half of full range for the A/D channels. Since the force transducer that provides CMD1 is not mechanically or electrically limited to ± 5 VDC, a software limit is provided to set CMD1 at either ± 5 VDC, as appropriate, when that value is exceeded. Output of the A/D converter is a 12 bit word, proportional to the voltage.

Error - The error module performs a double precision subtract of CMD1 from POS1 and sets computer gain through a series of shifts. It then determines polarity of the error and transfers to the appropriate output module.

Output - The output module sets countdown timers that establish the duration of the plus and minus portions of the PWM output signal. It switches DAC-1 and DAC-2 to -10 VDC when the "positive" counters have timed-out. When the "minus" counters time-out, it transfers control to the Input 2 module.

Input 2 - The Input 2 module controls the conversion of CMD2 and POS2 and provides limits on CMD2 in the same manner as Input 1. CMD2 and POS2 are for use in the Monitor functions.

Monitor - The Monitor module compares the redundant pilot command and rudder position input signals. If a difference in either of 1.5° is detected for a period of 0.128 seconds, the program is set to deenergize the DFBW holding relay and reverts control of the system into the ABU mode. The monitor also checks the magnitude of the error signal. If it exceeds 1.5° for 2 seconds, the DFBW holding relay is deenergized, and control of the system again reverts to the ABU mode. As long as the monitor does not detect an error, it transfers control back to the Input 1 module.

d. Flight Test Program Software

Support software, trade name "Microbug ROM" was purchased with the microcomputer equipment and enabled communications with the microcomputer via a Teletype Corp. Model 33TU teletype keyboard/printer reader/punch unit. An RS-232-TO-TTY adapter unit provided the interface through the Asynchronous Interface Adapter (ACIA), between the microcomputer and the teletype.

The communication consisted of entering both program and simulated input data, monitoring microcomputer operation, and dumping of programs onto paper tape for storage.

After the software modules were operating satisfactorily they were then merged to become an operational program. After checking the operational program the microcomputer integrated into the rudder system (in the laboratory), the program was then loaded into a PROM. The PROM was installed in the microcomputer and the operational program verified. All subsequent final system response testing and calibration for the flight configuration was performed with this PROM installed in the microcomputer.

A listing of the flight program software is contained in Appendix B of Reference 17. The program was designed to function at a rate of 500 HZ, and occupies 462 bytes of the available 4096 bytes of PROM and 18 bytes of 1024 bytes of "scratch pad" RAM. The PROM map is also shown.

e. Microcomputer Analog and Reduced Bit Resolution Program Software

In addition to the PROM software developed for the flight test program, additional software programs were developed to evaluate microcomputer performance as a function of reduced bit resolution and various PWM frequencies. These programs were designed for lab testing only, and were therefore stored on paper tape and entered into the microcomputer memory via the teletype.

f. Laboratory Tests

Laboratory tests were performed to integrate the microcomputer into the existing Phase V AFCAS system, evaluate system performance and compatibility with the microcomputer operational modes and software, and to ensure the equipment would function properly in the T-2C aircraft environment prior to installation in the aircraft.

The DFBW and ABU system operating modes were evaluated in the laboratory. In the ABU mode, the microcomputer is disconnected from the loop, and the aircraft sensors directly control the actuator through the EDU. This was the control used in Phase V of this program. Laboratory testing was also accomplished with the microcomputer supplying an analog signal instead of a PWM signal to the EDU. This testing was performed to obtain comparison data with the computer in the loop.

The actual aircraft hardware was used whenever possible in the lab test set-up to permit testing and evaluation of the flight hardware and to eliminate potential problems during subsequent installation and operation in the aircraft.

Included in the lab test set-up were the rudder LVDT feedback transducers, rudder actuator, EDU, microcomputer and associated power supply, and the control panel switches and control relay used in the aircraft. The 8000 psi (55 MPa) hydraulic pump, incorporated in the aircraft for the AFCAS installation, was not included in the lab set-up since the pump performed successfully in the previous Phase V of the AFCAS program and the potential benefits from including it in the lab set-up did not warrant the added expense. An 8000 psi (55 MPa) laboratory hydraulic pump was utilized for all tests requiring high pressure hydraulic flow.

A block diagram of the laboratory set-up is shown in Figure 94.

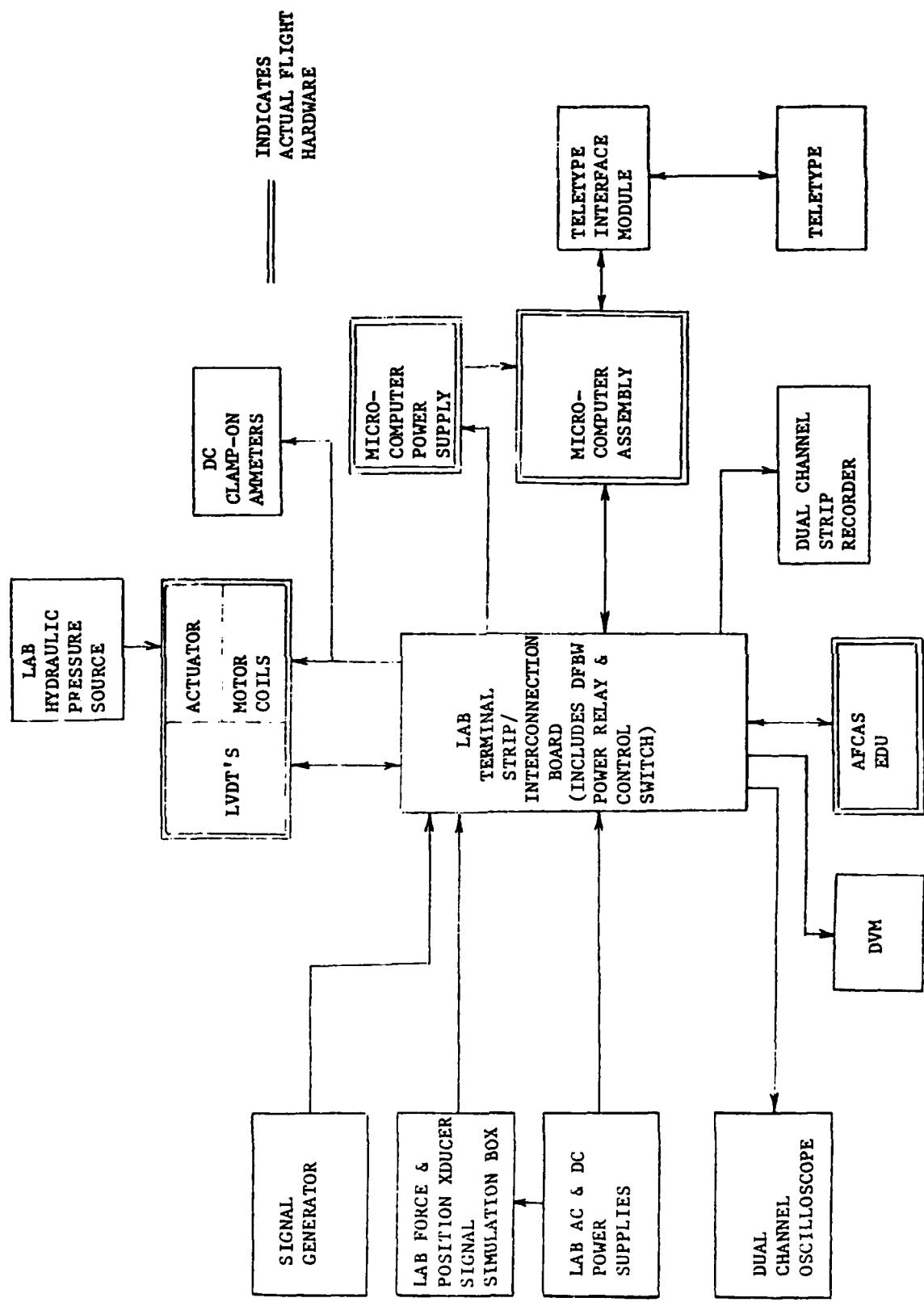


Figure 94. Laboratory Test Setup Block Diagram

3. Flight Test Programa. Flight Plan

The primary objective of the Flight Test Program was to evaluate performance of the computer-controlled direct digital fly-by-wire characteristics of the AFCAS in a T-2C aircraft. Approximately three flight hours were expected to be sufficient to evaluate performance, confirm prior analyses and laboratory tests, and provide a measure of confidence in system reliability.

The flight plan was designed to determine digital control characteristics at altitudes up to 30,000 feet (9.1 km) and speeds up to 340 knots (175 m/sec.) and to compare performance between the DFBW and ABU modes of operation.

Details of the flight plan are included in Appendix C of Reference 17.

b. Flight Test Results

Three flights were flown for a total of 4.7 hours. The pilot stated that performance of the Direct Digital Drive AFCAS test installation was completely satisfactory. No difference in "feel" was noted between the DFBW and ABU modes of operation. Additional pilot comments were similar to those listed in the Phase V AFCAS flight test program contained in Reference 16, and included:

- Directional Control Response was judged to be superior to the production T-2C.
- Pilot adaptation to "force control" of the rudder was quickly and easily acquired.
- The fixed pedals provide full rudder and allow much easier braking (in combination) without severe leg and foot extension that is required for conventional deflection controls.
- Hydraulic system fluid pressure and temperature were normal.
- No malfunctions occurred.

c. Flight Program Summary

Following is a summary of the Direct Digital Drive AFCAS flight program:

<u>FLIGHT</u>	<u>MAX. AIR SPEED & ALTITUDE</u>	<u>NZ</u>	<u>DURATION</u>
1	250 KOAS & 20,000 FT.	3.0"g"	1.4 Hours
2	250 KOAS & 20,000 FT.	3.0"g"	1.5 Hours
3	340 KOAS & 30,000 FT.	5.5"g"	1.8 Hours

ROCKWELL'S STUDIES OF PULSE MODULATED DIRECT DIGITAL DRIVE (D³) CONCEPTS

In advanced aircraft using digital fly-by-wire control, direct-drive actuators show considerable advantages because of their ruggedness, relative simplicity and high reliability. Both the Navy and Air Force have research programs under way to develop direct drive actuation.

Previous work on this project has demonstrated that low level analog signals, such as are provided by a D/A converter, can be suitably amplified into the high power forms required for closed loop control. Analog valve drivers using quadruple redundancy and optimized for linear (Class A) operation, were successfully tested and demonstrated in the laboratory. A modified PDP-11 minicomputer with

built-in A/D and D/A converters was used to close the loop both internal and external to the mini-computer. Flight worthy analog valve drivers were developed with IR&D funds. This design was fabricated and flown under Phase V of the AFCAS program.

Following Phase V, additional IR&D research was conducted to develop an approach that would (a) eliminate the need for D/A and A/D converters and (2) reduce the cooling requirements of the drive amplifiers.

Experiments were performed using error signals in various pulsed formats including "bang-bang" pulse width modulation and time dwell modulation.

The use of pulsed drive waveforms offers several potential advantages:

- D/A converters are not needed, since digital circuitry can generate the pulsed waveforms.
- A/D converters for "wrap-around" monitoring are not needed.
- Pulsed waveforms allow the valve driver amplifiers to act as switches rather than as linear amplifiers. As a result, the power dissipation in the drive transistors can be reduced by a factor of 10 or more, thereby improving the reliability and reducing the weight of the DFBW system is also improved as a result of the lower power dissipation.
- Dynamic pulsed waveforms are compatible with passive fault isolation schemes whereby a hardover computer output can be blocked without the need for a disconnect arrangement.

The results were sufficiently encouraging to warrant investigations into a new digital valve driver amplifier concept better suited to pulsed waveforms than are the linear analog drivers, and designed to exploit the potential advantages offered by the use of pulsed waveforms in aircraft having several actuation systems under direct digital control.

The digital drive concept, illustrated in Figure 95, was constructed and tested. The typical waveforms are illustrated in Figure 96. The circuit incorporates two features which reduce the size and weight over that of an analog drive. First, the transistors operate in a switching mode to reduce the internal power dissipation. Second, the circuit is designed to operate from a single polarity power supply.

In addition the circuit can be designed for dynamic operation, that is, a pulse rate is required to obtain an output. The absence of pulses results in zero output. If some form of AC coupling is provided between the computers and the power amplifier, a hardover failure of a computer output circuit would result in a passive failure and not a hardover failure. This permits the computers to perform self-monitoring and remove themselves from the line if a failure occurs. It does not matter if the computer output is zero or a plus voltage, it is still removed from the drive. This provides opportunities for many forms of redundancy in the actuator drive.

The breadboard digital driver was tested as a part of the closed loop system. A direct drive torque motor was used to provide a realistic load for the driver. A simulated actuator was configured to enable studying the dynamic response of the loop, which was closed by the computer. The computer generated the "surface error signal" in the pulse modulation format. The following specific results were achieved'

1. The circuit concept was verified and a data base was established for use in optimizing future operational circuit designs.

2. The closed loop test results showed that the desired closed loop frequency response can be obtained with proper compensation for the inductive characteristics of the valve coil.
3. The qualitative effects of pulsed waveforms on the torque motor were evaluated and found to have little effect on the motor when operated at frequencies above 500 Hz.

The closed loop frequency response without any form of compensation is shown in Figure 97. With computer compensation this bandwidth can be extended for small signals (1% of full travel) by a factor of 4 to about 26 Hz. This is illustrated in Figure 98.

For certain dynamic situations, large changes in the surface displacement would be desired requiring increased bandwidth for large signals.

Resistance in series with the motor coil is the simplest, most effective means of providing the needed compensation. The resultant large stroke response is shown in Figure 98. This method also offers the advantage of maintaining valve driver power bandwidth over the full range of error signal levels — that is, the large signal response is identical with the small signal response. The closed loop bandwidth will be limited only by the rate capacity of the actuator itself. The use of a series resistor provides an additional advantage of current-limiting protection in the event of a hardover failure.

This work, performed under company IR&D funds, demonstrated a method to take maximum advantage of pulse modulated control. While pulse modulation performs satisfactorily with analog (Class A) amplifiers, the use of drive circuits designed especially for pulse modulation produces a significant reduction in size and power (heating) dissipation in the drive circuit. Because of the low power dissipation of the electronics, the temperature rise is slight and the electronics could be packaged directly into the surface actuator. A direct drive actuator design that utilizes electronics of this type integrated into the actuator has been developed by Rockwell.

FLIGHT CONTROL COMPUTER COMPLEX – D³ ACTUATOR INTERFACE (obtained from Reference 18)

While the simplification of the surface actuator is significant, the major advantage of the direct drive concept is the reduced complexity of the computer-actuator interface. This is achieved because the direct drive concept provides methods of failure monitoring and redundancy management that are more compatible with digital technology. The direct drive torque motor is compatible with digital technology. The direct drive torque motor is compatible with digital (pulse modulation) control circuits. Pulse modulated signals eliminate the need for analog signal transmission and can be generated with digital timing circuits. Power amplification can be accomplished using highly efficient switching amplifiers. The direct digital drive provides several options in failure monitoring and redundancy management and can be used for triplex or quadruplex systems. It can also be used in conventional redundancy management systems and is suitable for a fault tolerant design.

1. Pulse Modulation Command

The use of an electromechanical force motor to power the main control valve requires electrical power amplification to replace the hydraulic power amplification. While servo power amplifiers, capable of handling the power requirement (25 to 50 watts) are readily available, the characteristics of the force motor are advantageous for other methods of power amplification. The force motors are limited travel devices with the windings having fixed resistance and inductance typical of electromagnetic coils rather than the characteristics of a motor. Utilizing the R/L time constant of the coil winding as a filter makes the force motor well suited to the use of pulse modulated techniques.

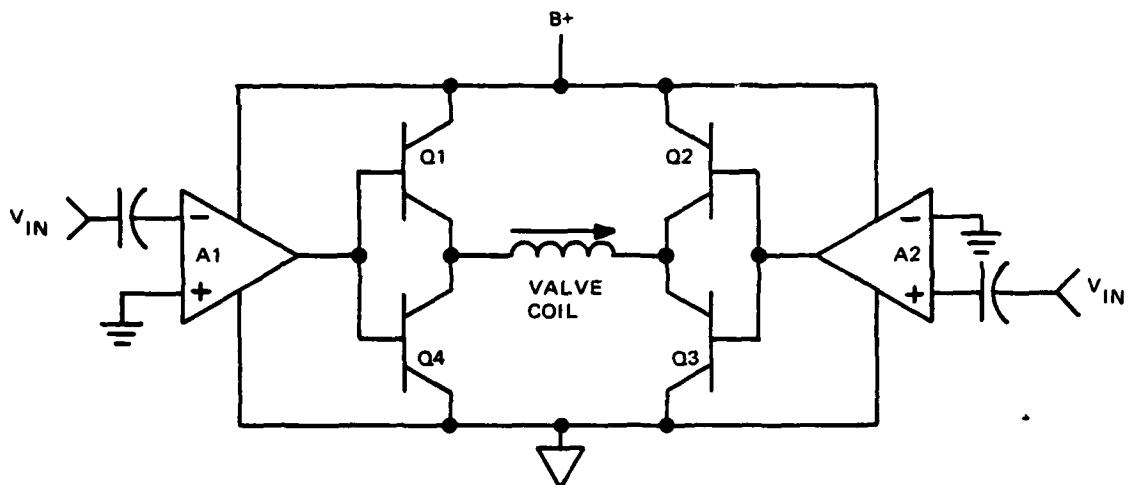


Figure 95. Digital Drive Unit Concept

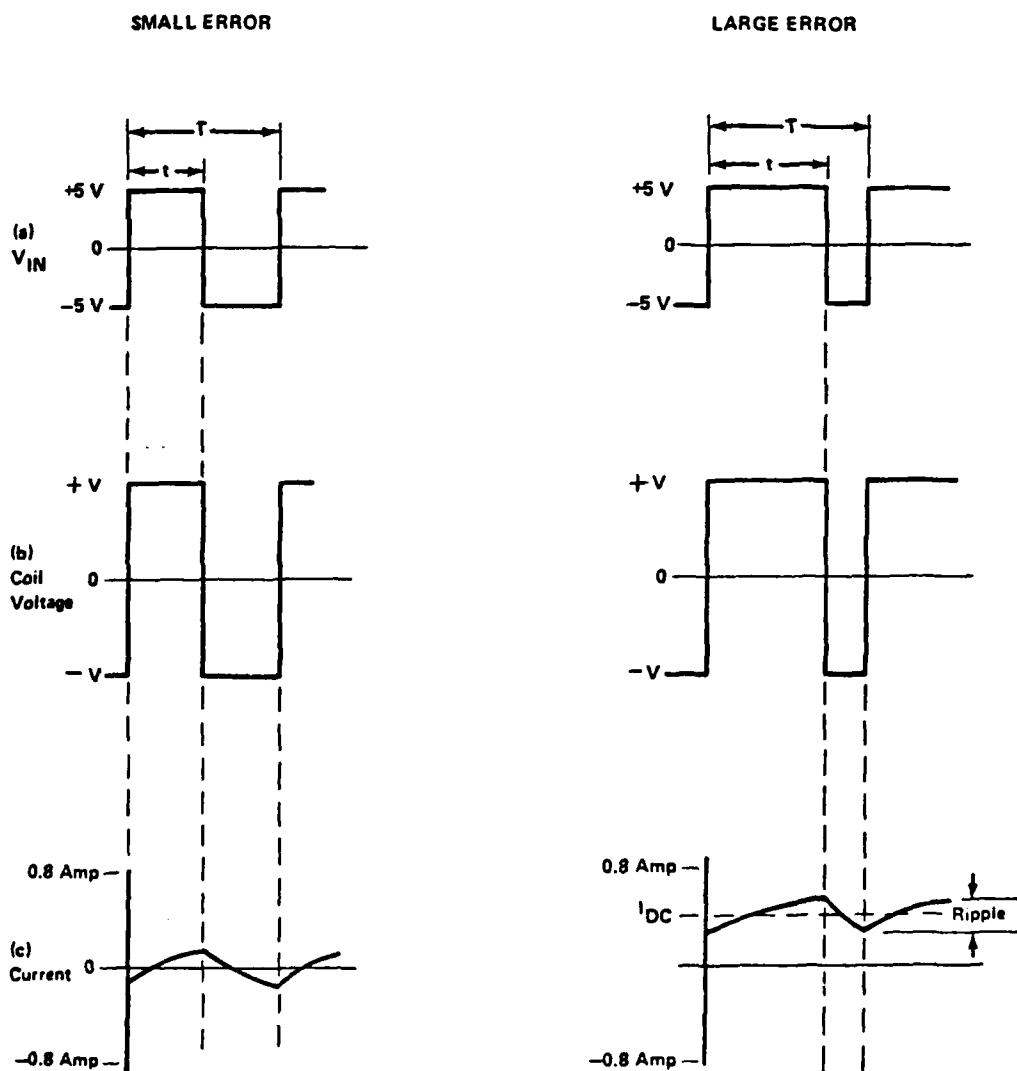


Figure 96. Digital Drive Concept Waveforms

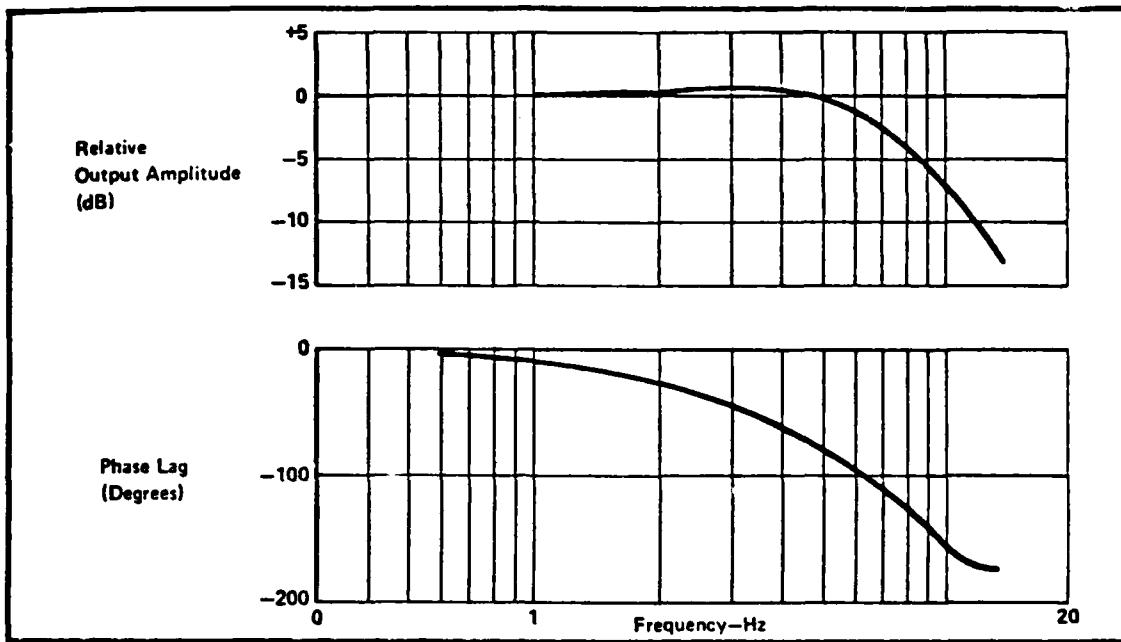


Figure 97. Closed Loop Response (Uncompensated)

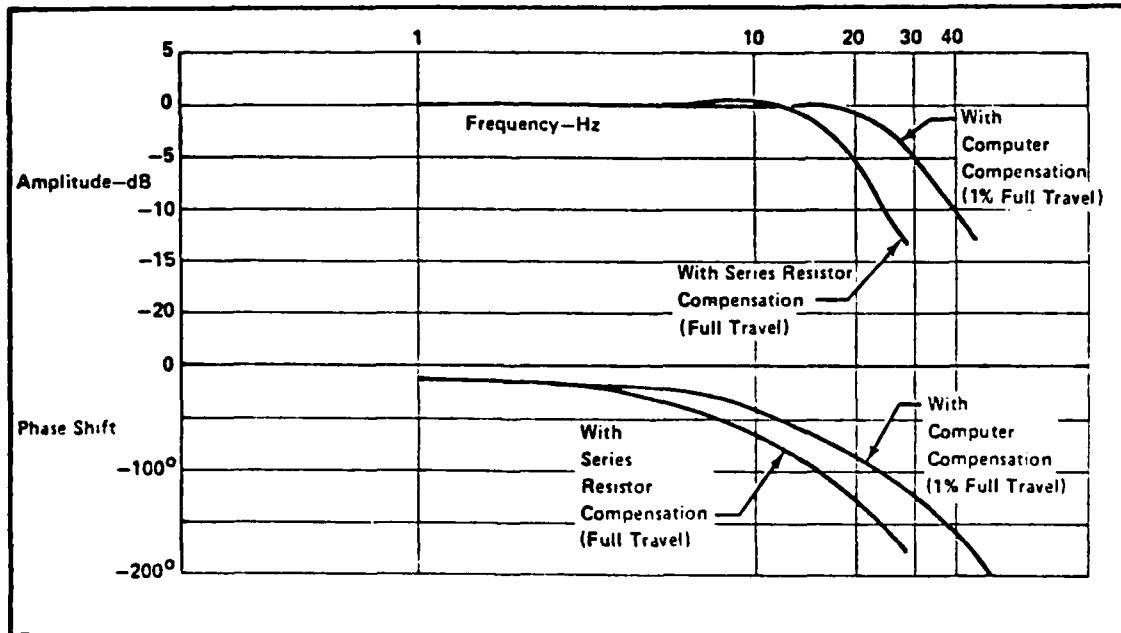


Figure 98. Effect of Compensation Method on Closed Loop Response

Pulse modulation signals can be generated with simple digital timing circuits and the actual power amplification can be accomplished in switching circuits, eliminating the need for analog servo amplifiers.

Table 9 lists the characteristics of the torque motors used in the AFCAS program. Several forms of pulse modulation have been evaluated including pulse width modulation (PWM), time dwell modulation, and an on-off or "bang-bang" control. While all three types of modulation are usable, pulse width modulation proved superior for small signals, offering the best response and linearity. This occurs because PWM uses a constant carrier frequency with the signals producing only slight changes in the plus and minus portions of the wave. Other forms of pulse modulation require a much wider bandwidth to accommodate the frequency variations.

The modulation rate must be selected to meet the requirements of the electromechanical force motor. The rate of modulation must be greater than the natural mechanical frequency of the force motor and also well in excess of the R/L time constant of the motor coils. When these requirements are met, the characteristic wave shapes shown in Figure 99 will be present at the motor. While the signal voltage to the switching amplifier will be pulsed, the current through the force motor windings will be well filtered by the inductance of the coils.

TABLE 9 – ELECTROMECHANICAL FORCE MOTOR DATA

COILS PER MOTOR	—	4
VOLTS PER COIL	—	10V
CURRENT PER COIL	—	0.8A (RATED TORQUE)
COIL RESISTANCE	—	10 OHMS
COIL INDUCTANCE	—	0.6 H.
POWER PER COIL	—	8 WATTS (RATED TORQUE)
NATURAL FREQUENCY	—	230 Hz
SHEAR OUT FORCE	—	80 LBS (RATED)

The principle of the switching circuit is quite simple as illustrated in the schematic of Figure 95. There is one switching amplifier for each channel of redundancy. This form of switching amplifier offers several advantages. By using four switches operating in odd or even pairs (Q1 & Q3 or Q2 & Q4) a single switch failure will not cause a hardover coil command. However, a single switch failure will cause excessive current as half the cycle will produce a shorted supply, thus making current monitoring a positive fault indicator.

Signal amplifiers A1 and A2 are used to buffer the input signals. A failure in either of these stages cannot cause a hardover command, but will result in a bias current in the switching amplifier that can be readily detected and the power source subsequently turned off. If the pulse modulated input signal is AC coupled, pulses are required for the signal to be transmitted to the switching amplifier. A lack of pulses results in a fail passive circuit, even if the command voltage were to remain in a full on condition.

The pulse modulated command signals can be generated by the computer if cycle time is not a concern. The pulse modulated signal can also be generated by simple timing circuits consisting of several logic gates and a counter. The exact design will vary with the application. The Motorola timer MC6840 is one method of generating a PWM signal that is very flexible for test work. This

chip is under computer control and the modulation characteristics can be readily varied. In flight tests performed at Rockwell using the direct drive actuator in the rudder system of the T-2C aircraft, the pulse modulated signal was generated by the computer. For these test programs, closing the loop through the computer had advantages for monitoring system operation, and computer cycle time was not a concern.

The switching amplifier that produces the main power amplification can be located at the computer, the actuator, or anywhere in between. The most desirable location will usually be at or in the actuator. This eliminates the need for transmission of analog power signals and permits power monitoring to verify the current used at the electromechanical force motor rather than on the transmission side of the drive. The approach requires two additional wires per stage to transmit power to the actuator.

2. D³ Redundancy Concept

The redundancy concept shown in Figure 100 for a quadruplex direct drive actuation is very similar to existing redundancy concepts. The dual tandem actuator employs two electromechanical force motors driving the main control valve. Each force motor contains four windings. Each of four switching amplifiers drive two windings in series, one from each force motor. The switching amplifiers are driven by a PWM signal from four separate computers, eliminating the need to transmit analog information. As the switching amplifier is driven into saturation, neither the quality or exact DC level of the power source are critical and will not effect loop gain. The power (current) supplied to each switching amplifier is monitored at the source. When improper power measurements are observed, the power source can be switched off. It is assumed that the aircraft electrical power concept for the drive circuits will be the same as for the computers.

The switching amplifiers operate very efficiently with the switches either at zero voltage drop or zero current. The power dissipation in the amplifier is no greater than the present class A servo amplifiers used to power the electrohydraulic valves. The switching amplifiers are not gain sensitive, and temperature effects on components will not affect system operation. This makes it practical to locate the switching circuits in the higher temperature environment of the actuator. Locating the switching amplifier at the actuator permits transmitting non-critical pulse modulated signals and non-critical power to the actuator. The command signal is now a digital signal instead of a low level analog signal.

Since the pulse modulated command signals are generated by timing circuits, it is possible to monitor the command signals to a greater precision than possible with an analog drive. This enhances the monitoring of the computer outputs and allows the computers to increase the accuracy of self test. Computer redundancy requirements become less critical with improved output monitoring which should make the triplex system very attractive to D³ actuation.

The redundancy concept shown in Figure 100 is functionally quite similar to present systems. The only significant redundancy difference is the location of the force monitor and channel turn-off switches at the computer. In present systems multiple hydraulic force monitors and solenoids must be located in the actuator environment. This greatly increases the number of wires required between the actuator and computer. Present systems need 80 to 100 wires compared to approximately 30 for the D³ system.

3. Fault Tolerant Redundancy

In the direct drive system presently under test, loop gains in excess of 90 (15 Hz) are possible because of the excellent dynamics of the direct drive torque motor. With this high loop gain, a hard-over failure of one of the four channels will cause a bias shift at the actuator of less than 1% of its

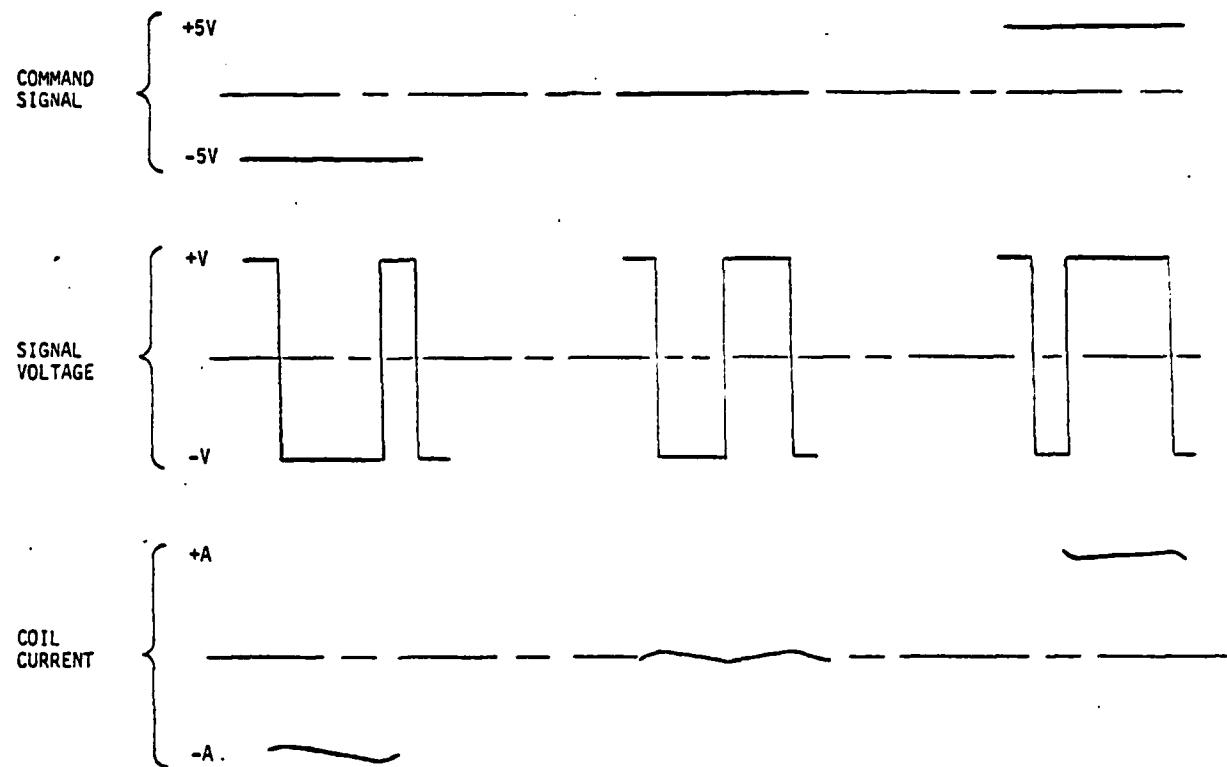


Figure 99. Pulse Modulated Waveforms

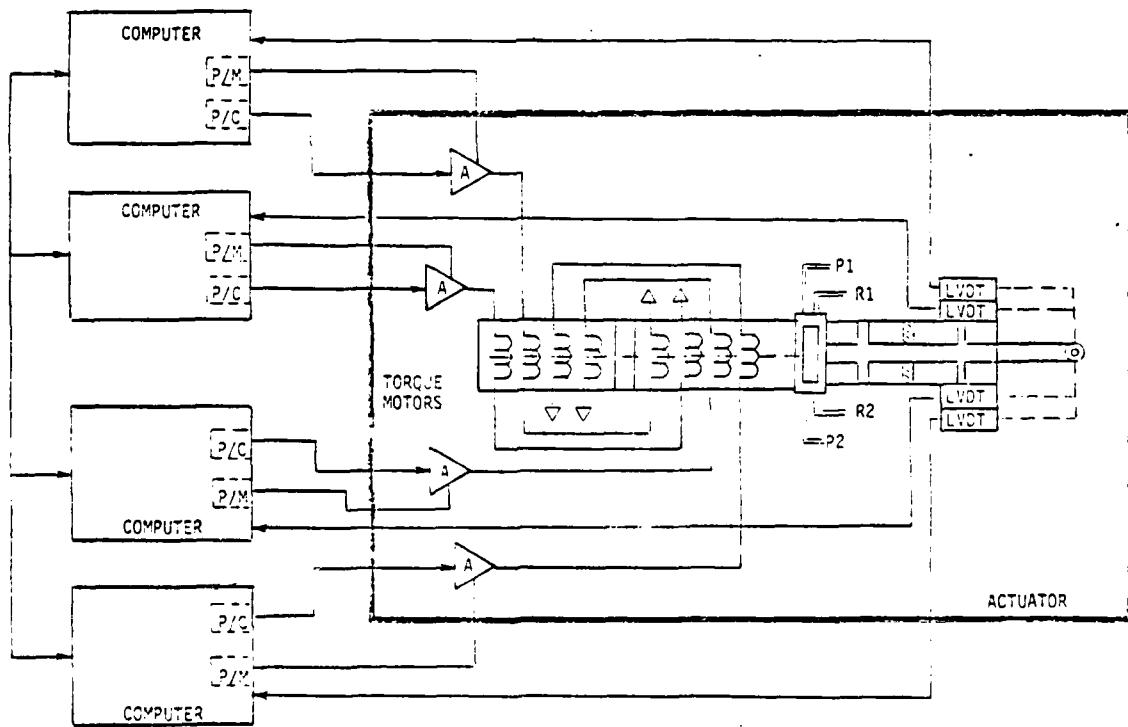


Figure 100. D3 Redundancy Concept

total travel. In some applications, surface responses of 15 Hz are not desirable. The use of lower loop gains will cause a corresponding increase in the bias error due to a single channel failure.

One approach to reduce the bias error of a failure is to use four drivers per actuator stage (eight per dual actuator). This will reduce the bias error of a single "hardover" failure by more than a factor of 2 and will allow the unit to continue operation with up to 3 identical "hardover" failures in the drive electronics. With this approach, current monitoring and switching would not be required. The simplicity of the switching amplifier makes this a reasonable approach. The use of an interloop feedback (monitoring force motor position) is also a valid approach to reducing effects of hardover failures, but adds considerable complexity to the overall interface.

As shown earlier, single failures of the switching circuit, while not producing "hardover" failures, can result in drawing excessive current from the power source. Therefore, regardless of which redundancy method is selected, "short circuit" protection of the power source must be provided. This protection also removes power from a failed channel. The actual redundancy method selected will depend on the system application.

This system approach simplifies the actuation, the command electronics, and the interface by increasing the use of digital electronics. The only area that remains analog is the feedback sensors. A program is presently underway that will remove the analog sensors and reduce the electronic hazards such as EMI, EMP, lightning, high energy radiation, and spurious signals.

THE HYDRA-OPTIC FLIGHT CONTROL ACTUATION SYSTEM (HOFCAS) DEVELOPMENT PROGRAM

The AFCAS concept developed by the Naval Air Development Center (NADC) is a direct-drive, lightweight, hydraulic surface actuation system capable of being controlled with a direct digital command. This Direct Digital Drive (D³) actuation system and the lightweight hydraulics, have been successfully flight tested and shown to be a practical approach to surface actuation. This project utilizes AFCAS with new approaches incorporated to eliminate the conventional interconnecting cabling to the surface actuator.

A major concern for future fly-by-wire aircraft is protection against electronic hazards such as EMI, EMP, lightning, high energy radiation, and spurious signals. If control signals can be transmitted via light pulses, particularly to the actuation regions, then electromagnetic transient protection can be greatly enhanced. By removing all electrical connections to the actuator, electromagnetic coupling which would allow RFI, lightning or other transient signals to enter would be eliminated. The electrical power required to receive and utilize the light pulses would be generated hydraulically within the actuator. The electronics to restore the digital signal and drive the electrohydraulic valve would also be contained within or on the actuator. The system developed for this project offers an approach for meeting the goal of a direct drive surface actuator with no interconnecting cabling.

The AFCAS program that preceded this study was a six phase program that explored the feasibility of direct drive, developed the concept, and evaluated it in a series of laboratory and flight test programs. Phase VI of this program incorporated the microcomputer and successfully demonstrated the D³ capability of the AFCAS. Pulse modulated signals, generated in the microcomputer, were power amplified in an Electronic Drive Unit (EDU) and used to control the direct drive actuator. An analog backup capability was also provided, whereby the EDU controlled the actuator directly and the microcomputer was not in the control loop.

The general objective of this program was to assemble and test a direct-drive, lightweight, hydraulic surface actuator system that provides immunity to all forms of electromagnetic radiation. The specific goal was to operate the digital AFCAS without the need for electrical power being transmitted to the actuator. This was accomplished by modifying the AFCAS to operate with a hydraulically-driven alternator supplying the electrical power, and adding an optical command link between the microcomputer and the direct drive actuator electronics.

TECHNICAL APPROACH

The technical approach used to develop the Hydra-Optic Flight Control Actuation System (HOFCAS) was to utilize assets developed under the previous six-phase AFCAS program and to demonstrate, through laboratory testing, that the HOFCAS concept is suitable for flight in the T-2C Demonstrator Aircraft. The AFCAS actuator, the transducers, and the microcomputer power supply were used without change. The microcomputer and the EDU were modified to incorporate optical data transmission. A Hydra-Powered Alternator (HPA) was added to supply the electrical power needed for the actuator drive unit. Additional changes were made to the system switching and interconnecting wiring to incorporate the new modes of operation.

The system configuration was designed and tested to verify that the HOFCAS is suitable for controlling the rudder of a T-2C aircraft, shown in Figure 84. For this application the T-2C control system will be modified from a mechanically powered cable system to a full powered Digital Fly-By-Light (DFBL) system with an Analog Back-Up (ABU) mode of operation. The original cable system between the rudder pedals and the rudder will be removed. The rudder pedal will be attached to force transducers. Force on the rudder pedal is converted to a proportional electrical signal by these force transducers. This signal is supplied to a microcomputer where it is summed with the feedback signal and processed into a pulse modulated error signal. The pulse modulated error signal is transmitted optically through fiber optic cables to the EDU where it is converted to electrical signals, power amplified, and used to control the torque motor of a direct drive hydraulic rudder actuator. This modified system provided full power DFBL control of the rudder, instead of the manually operated rudder of the basic T-2C aircraft.

The HOFCAS mounting requirements, lead length, and hydraulic and electrical power sources were selected to be compatible with the T-2C. The HOFCAS evaluation criteria were selected to meet the dynamic performance requirement of the T-2C aircraft. Safety provisions, including failure mode evaluation, were based on T-2C flight safety requirements. The previous AFCAS flight test program included a hydraulic bypass valve on the direct drive actuator. This device allows the rudder to seek a trail position if the rudder control system were lost. Flight testing has established the aircraft can be safely landed with the rudder in the trail position. This feature establishes a third level of redundancy beyond the DFBL and the ABU modes of control.

All the major components needed to fly the HOFCAS in the T-2C were assembled in the laboratory for the system testing. System and component tests were performed to insure satisfactory operation in the T-2C. Additional system tests were performed that exceeded T-2C requirements. These tests were performed to establish the performance levels possible with the present HOFCAS and to identify areas where improvements could be made if necessary.

This approach makes use of existing Navy hardware to test and evaluate several critical portions of an actuation system that eliminates the need for wiring to transmit power and signal to the actuation region. The Navy currently has a separate program for developing an optical position sensor to provide a feedback for this direct drive hardware. After the optical sensor, the HPA, and the EDU are integrated into a common actuation package with the D³ actuator, the general objectives of this program will be achieved.

DESCRIPTION OF THE HOFCAS

The HOFCAS utilizes a direct drive actuation system, commanded by light, with self contained electrical power to control an aircraft rudder actuation system. The system hardware for this program is a modification of the Phase VI Direct Digital Drive AFCAS program hardware previously evaluated and described in Reference 17.

A simplified block diagram showing the major components of the HOFCAS system is contained in Figure 101. The shaded areas in Figure 101 indicate the units modified or added for the HOFCAS. The microcomputer and the EDU were modified to incorporate the optical command link (new) and the HOFCAS Hydra-Powered Alternator (HPA) is a new unit added to provide electrical power for the EDU in the Digital Fly-By-Light (DFBL) mode. Two modes of system operation are provided, the DFBL and the ABU modes. Figure 102 contains a block diagram that illustrates the two modes of operation.

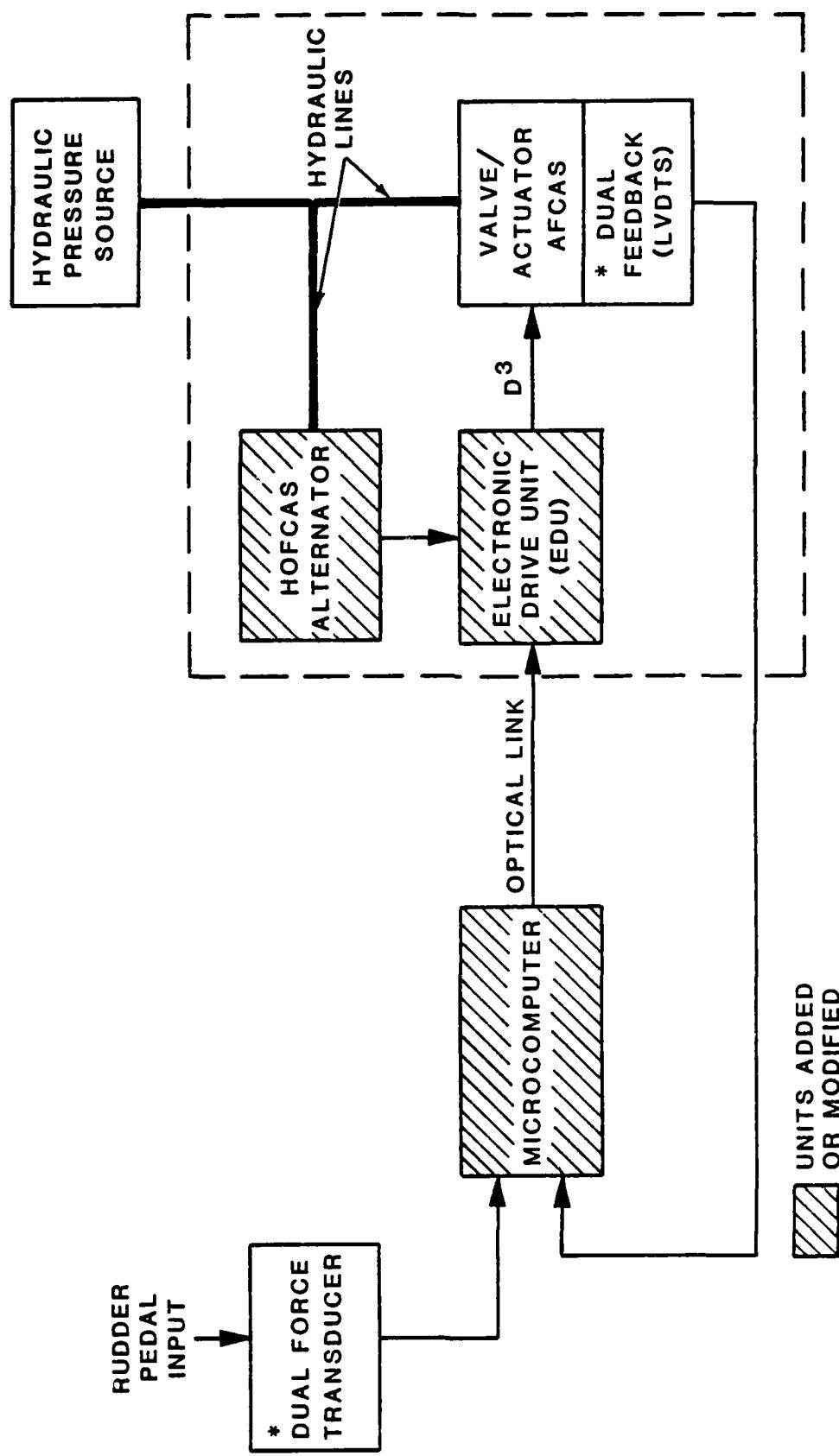


Figure 101. HOFCAS Simplified Block Diagram

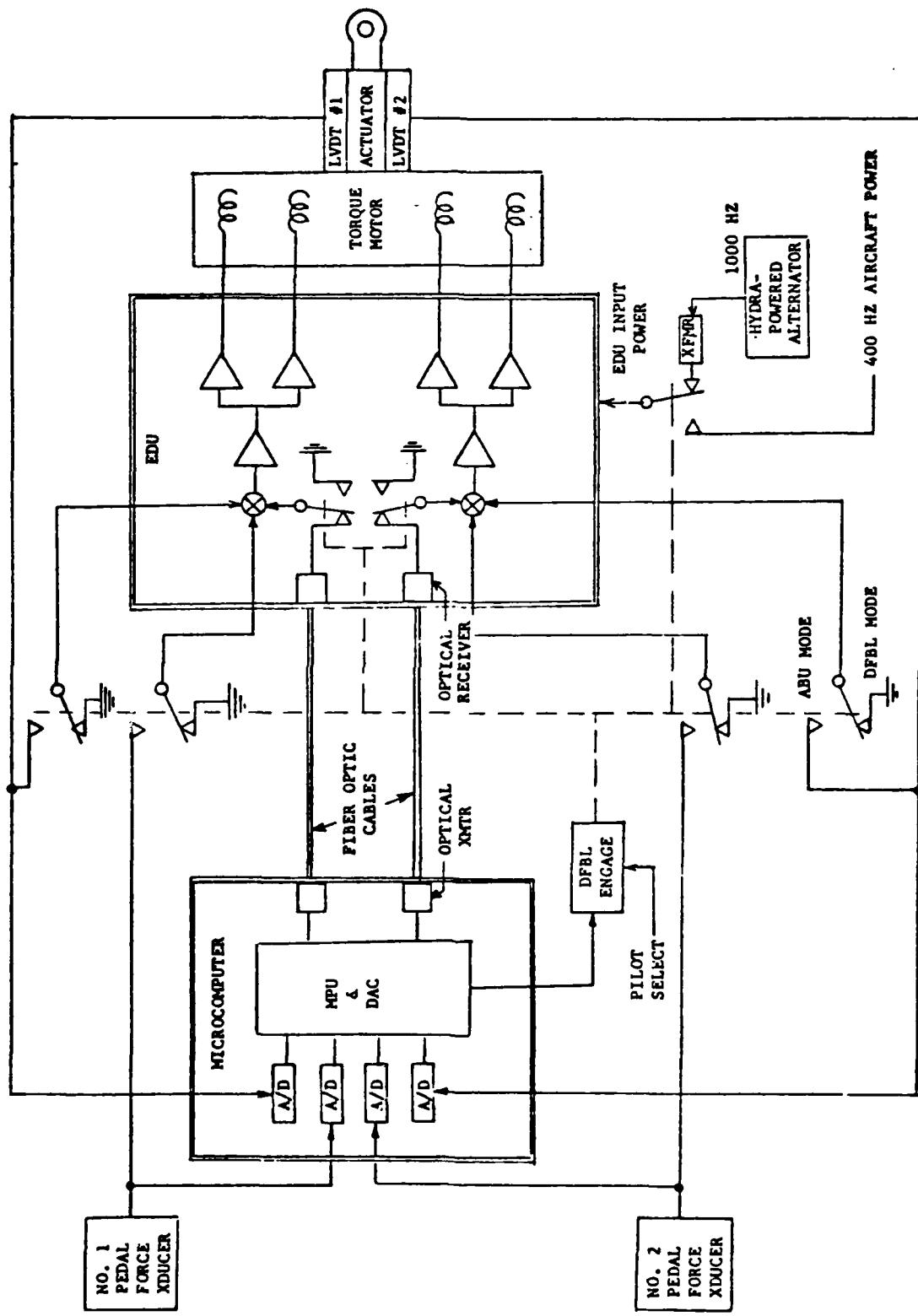


Figure 102. HOFCAS Operational Modes, Simplified Block Diagram

The DFBL mode is selected by momentarily holding the cockpit DFBL ENGAGE switch to ON, energizing the DFBL control relays and resulting in the following:

- The pedal command and actuator feedback transducer outputs are connected to the microcomputer input.
- The microcomputer output is connected to the EDU via the fiber optic control link.
- Hydraulic pressure is supplied to the HOFCAS HPA, and the 26 VAC, 1000 Hz HPA output is used to power the EDU via a 26 VAC to 115 VAC step-up transformer. The EDU input power is utilized to provide excitation for the transducers and to condition, amplify, and convert the optical signals into electrical signals which are power amplified and used to drive the torque motor coils.

If the microcomputer senses the system is functioning properly, a power ground is continuously supplied to the DFBL ENGAGE switch holding coil by the microcomputer and the system remains in the DFBL mode.

The ABU mode can be selected by manually placing the DFBL ENGAGE switch to OFF, or automatically by the microcomputer removing the power ground from the DFBL ENGAGE switch holding coil if abnormal system operating errors occur.

Switching to the ABU mode results in the following:

- The fiber optic control link output from the microcomputer is disconnected from the EDU, and the pedal command and actuator feedback transducer outputs are connected directly to the EDU where they are summed and amplified to drive the four torque motor coils of the actuator direct drive valve.
- Hydraulic pressure is removed from the HOFCAS HPA, and the EDU is powered by 115 VAC, 400 Hz aircraft power.

The various system components are discussed in the following paragraphs.

1. Hydra-Powered Alternator (HPA)

The HOFCAS HPA assembly, Berteia P/N 287700, supplies electrical power to the EDU in the DFBL mode. The alternator rated output, at 3000 psi (21 MPa) Δ P and a fluid flow of 0.72 gpm (2.7 L/min), is 25 watts @ 26 VAC, 1000 Hz. The HPA was not used in the previous Phase VI of the Digital AFCAS program and has been added for HOFCAS.

The unit consists of a hydraulically driven 2" (5.1 cm) diameter turbine wheel coupled to a conventional permanent magnet electric alternator, spun by fluid discharged from a .020" (.05 cm) diameter orifice. The 6-pole configuration of the alternator, Electro-Kinetics Model No. 4922-2, provides a frequency of 1000 Hz at 10,000 rpm.

The alternator output is nearly sinusoidal with a resistance load, and distorts to a pseudo square wave when inductive loads are applied, (detailed data describing alternator output and waveforms are contained in section 3.3.1 of Reference 19).

The HPA turbine wheel and alternator are shown in Figure 103 and the HPA assembly is shown in Figure 104. Additional HPA data is supplied in Appendix B of Reference 19.

2. Electronic Drive Unit (EDU)

The EDU contains the electronics required for converting optical inputs to electrical input signals, signal conditioning and amplification, signal summation, and power amplification to current drive the actuator torque motor. The unit was designed by the North American Aircraft Division (NAAD) and contains two independent channels, each subdivided into dual valve driver circuits.

Each of the dual valve driver circuits employ current feedback with a highly reliable "Darlington" power transistor configuration. Redundant power supplies are used. The circuitry is designed so that in the event an output stage fails "hardover", the voltage applied to a motor coil will not exceed its rated value. This limiting feature permits a sub-unit failure to be compensated or nullified by another subunit. Closed loop tests showed that operation of the redundant subunits provided high immunity to component failures.

The dual channel fiber optic receiver amplifier added for HOFCAS converts the microcomputer optical output signals to electrical input signals for the EDU. Each receiver amplifier channel contains a detector, two stages of gain, and a passive isolation circuit. The optical signals received consist of PWM information, and the passive isolation feature prevents a hardover of the rudder actuator if the drive signal is lost due to a microcomputer or fiber optic loop failure.

The dual channel fiber optic receiver amplifier and associated detectors are located on a single module board, contained in a housing mounted on the EDU. All electrical and optical connections to the module board were made to permit easy removal if return of the EDU to its original configuration is desired.

The modified EDU with fiber optic cables is depicted in Figure 105 and is functional schematic, including the fiber optic receiver amplifier, is included in Figure 2-3 of Reference 19.

3. Microcomputer Assembly and Power Supply

The microcomputer assembly is housed in an enclosed unit, and consists of the following subassemblies:

- Motorola Monoboard Microcomputer Module
- Analog-To-Digital (A/D) Converter Module
- Digital-To-Analog (D/A) Converter Module
- Card Cage and Mother Board Assembly
- Signal Conditioning Board
- Fiber Optics Transmitter Board Assembly

The microcomputer module is a complete computer-on-a-board having all the processing and control required for a microcomputer-based system. It incorporates the MC 6800 MPU, 1 K of Random Access Memory (RAM), provisions for 4 K of Programmable Read Only Memory (PROM), timing and control, buffers, an Asynchronous Interface Adapter (ACIA) and two Peripheral Interface Adapters (PIA).



Figure 103. HPA Turbine Wheel and Alternator

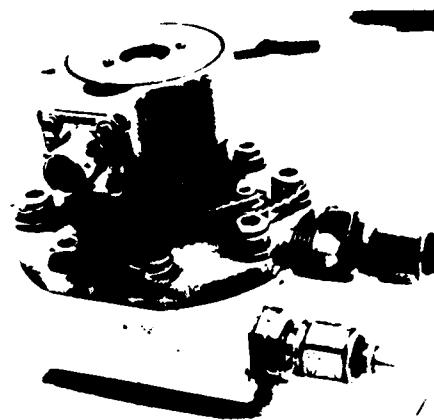


Figure 104. HPA Assembly

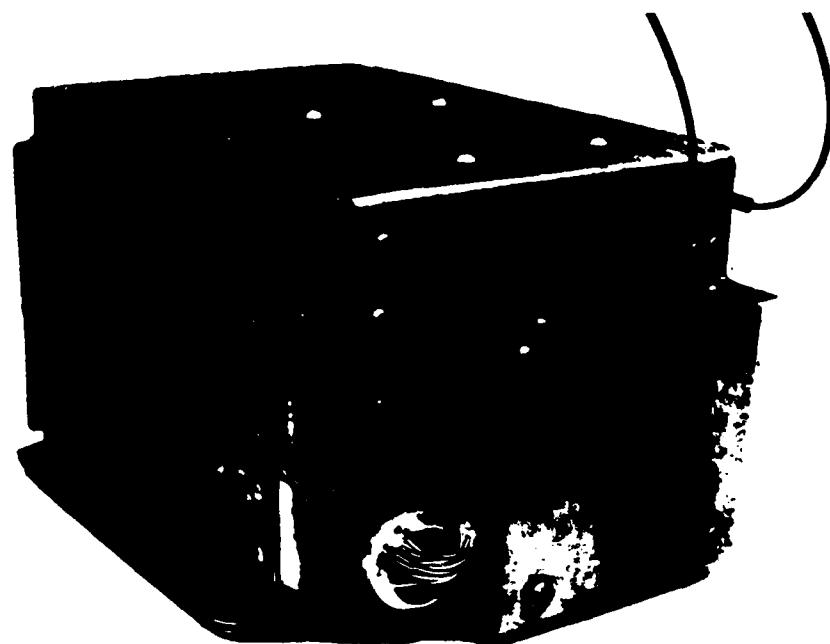


Figure 105. EDU With Fiber Optic Cables Attached

The A/D converter module consists of eight channels of A/D conversion of which four are utilized. The D/A converter module consists of four channels of D/A conversion of which three are utilized.

The signal conditioning board contains four channels of sensor signal conditioning and a relay driver that interfaces the microcomputer monitor output with the system control logic.

The dual channel fiber optic transmitter board was added for HOFCAS. Located in the microcomputer housing, the board contains a fiber optic photo diode and dropping resistor for each channel, to convert the microcomputer PWM output to optical form for transmission to the EDU. Two fiber optic cables, each 10 feet (3.05 m) long, interconnect the microcomputer and EDU. The 10 foot (3.05 m) cables simulate the length required to connect an actuator, with its electronic EDU Located at the rudder, to the microcomputer located in the equipment bay. The microcomputer unit with the fiber optic cables attached is shown in Figure 106.

A separate power supply (Motorola P/N M68MMPS1) is provided to supply power to the microcomputer. The power supply converts single-phase, 115 VAC, 400 Hz to + 5 VDC and \pm 12 VDC.

Additional information on the microcomputer is contained in Appendix A of Reference 19.

The software of the microcomputer is the same as that used in the flight testing of the D³ AFCAS concept (Reference (7)).



Figure 106. Microcomputer Unit With Fiber Optic Cables Attached

LABORATORY AND INTEGRATION TESTS

Laboratory tests were accomplished to integrate the HOFCAS with the existing Phase VI digital AFCAS system, perform system functional tests, evaluate system performance and compatibility with the HOFCAS modification, and to determine the effects of the HOFCAS modification on system performance. Temperature tests were also performed to verify the equipment would function properly in the T-2C aircraft operating environment.

Laboratory tests were accomplished using the actual aircraft components to simulate flight conditions in preparation for flight testing of the system at a later date.

Included in the lab setup were the rudder LVDT feedback transducers, rudder actuator, EDU, microcomputer and associated power supply, HPA, fiber optic link including emitters, detectors and fiber optic cables, and the switches and control relays used in the aircraft.

The lab wire harness was configured to simulate the aircraft wiring, and a terminal strip/interconnection board provided control, test points, and the interface between the wire harness, system components, and lab test equipment. The Lab Signal Simulator Box was included as part of the lab test setup, and provided simulated pedal force commands, LVDT feedback signals, and microcomputer PWM output signals during certain phases of system testing. The two fiber optic cables, connecting the microcomputer output to the EDU, were each 10 feet (3.05 m) in length simulating the actual aircraft cable length requirements. A portable lab hydraulic pressure source was used for all lab testing.

The bandpass requirements and therefore the system loop gain were established by T-2C aircraft dynamic requirements for the integration tests. System dynamic response tests were performed to demonstrate that the HOFCAS operational modes meet these dynamic requirements and are satisfactory for controlling the rudder of the T-2C aircraft. The lab test setup was configured to provide either HPA 1000 HZ or simulated aircraft 400 HZ power for the EDU in the HOFCAS DFBL and ABU modes, and a DFBW mode of operation. A function generator provided sinusoidal and step input signals to the system force transducer inputs and a 2-channel strip chart recorder was used to record the function generator output and rudder actuator feedback signal.

ANALYSIS OF TEST RESULTS

The test results demonstrate that the HOFCAS is satisfactory for operation in the rudder system of the T-2C aircraft. The testing was divided into three categories. The first group of tests accomplished the functional testing of the added system components, especially the HPA. The second group of tests evaluated the HOFCAS for operation with the T-2C rudder. The third set of tests established performance limits and provided data for suggested improvements in the HOFCAS concept.

1. Hydra-Powered Alternator (HPA)

The functional tests established the HPA operational capability and demonstrated the unit is adequate to supply the electrical power needed to operate the rudder actuator of a T-2C aircraft. The data obtained has the characteristics expected for the HPA design. The design is described in Appendix B of Reference 19.

The speed (frequency) and the no load voltage of the alternator are directly related to hydraulic flow, which is controlled by the turbine nozzle diameter and the applied hydraulic pressure. Increased flow through the fixed nozzle results in increased losses and additional heating of the hydraulic fluid.

The alternator used for this project was an adaptation of an existing missile power source and is wound to provide alternator protection against overloads. This characteristic becomes apparent in the tests data. As the output current is increased, the voltage decreases and total power output remains the same. At the same time, the unit shows very little change in speed (frequency) with increased current. As the power output remains constant at rated load, it is apparent the hydraulic turbine load is constant and the speed would not vary. In addition, the HPA has a very low efficiency so that slight variations in extracted power have very little effect on total torque required to drive the alternator.

Wave shapes of the alternator output showed the waveform with a resistive load to be nearly sinusoidal. For this system application the output power is supplied to power supplies in the EDU through a transformer and the wave shape is not critical to system operation. When the HPA was used to power the EDU, the load became highly inductive and the wave shape approached a square wave. While this does not directly affect system operation, the distorted wave shape reduces transformer and power supply efficiencies and further increases the loading effects on the alternator.

The HPA used for this project is satisfactory for flight testing, and will demonstrate the concept of a D³ actuation system without electrical power and signals routed to the actuation surface. Several areas of improvement in the HPA performance are desirable and are discussed in Appendix B of Reference 19.

2. Fiber Optic Link

The fiber optic link performed satisfactorily as expected. No difference in system performance was noted between the fiber optic link or direct wiring method of connecting the microcomputer output to the EDU.

3. Flight Test Configuration

HOFCAS integration tests were performed using the hardware configured to the T-2C aircraft installation. These tests consisted primarily of dynamic response and failure mode evaluation. Frequency response tests showed a slight reduction in system performance with the EDU powered by the HPA 1000 Hz output compared to 400 Hz power.

The slight reduction in dynamic response, when the EDU is powered by the HPA 1000 Hz output, is due to the limited power capability of the HPA which results in a voltage drop at the torque motor coils when additional power is requested. The voltage drop causes a reduction in loop gain and a corresponding loss of bandpass.

The HPA used in this project is capable of a steady state 26 watt output with no capability of handling higher power demand transients. The EDU peak power requirement is 50 watts, consisting of 32 watts for the torque motor and 18 watts for EDU amplifiers, power supplies, and transducer excitation. The torque motor characteristics (see Figure 107) are such that at 26 watts total power (8 watts at the torque motor) the major portion of the spool travel has occurred and the rated flow of the valve is already obtained.

The additional power requirement to the torque motor is needed to obtain the full particle shear out capability and is not required for full hydraulic flow. For the T-2C application, if the full torque motor force is required, the system will revert to the ABU mode with aircraft power and full shear force will be available.

The step response characteristics determined during the tests again show the effects of limited power to the EDU. This effect is more pronounced with step demands as they require high power peaks.

4. HOFCAS Performance

Results of the dynamic tests performed in this project successfully demonstrated the suitability for HOFCAS operation in a T-2C aircraft and provided additional data to demonstrate the potential of a HOFCAS design.

The performance of the AFCAS actuator was evaluated in a previous flight test program. The measured frequency response shown in Figure 108 is approximately 13 Hz. This response was considerably above that required for the flight test program.

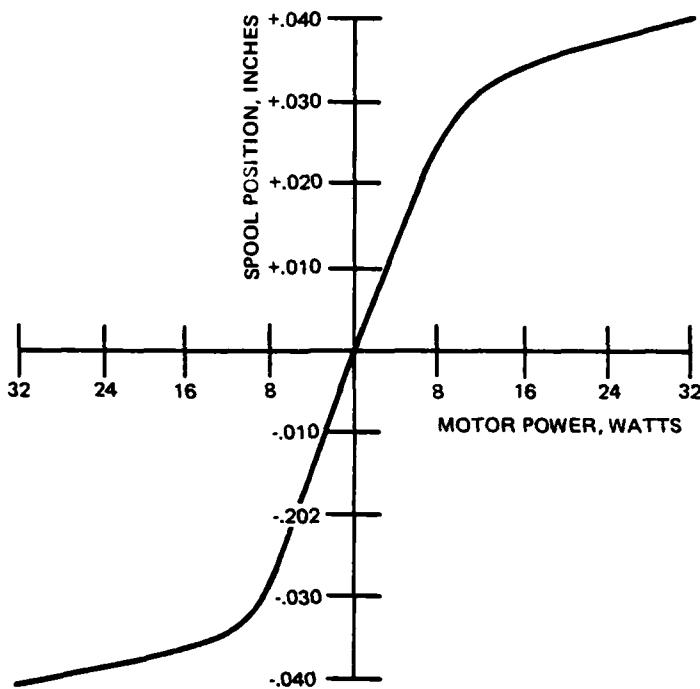


Figure 107. Torque Motor Characteristics

The D³ operation of this actuator used the same analog EDU to drive the torque motor of the actuator. Since the EDU was not optimized for digital drive some bandpass was lost, and the response is that shown on Figure 109 for aircraft power. Rockwell laboratory tests of a circuit optimized for direct digital drive have produced the frequency response (25 Hz) shown in Figure 110. As the natural frequency of the torque motor is approximately 230 Hz, the dynamic response is limited either by the valve design or the electronic drive design.

The data of Figure 109 shows the dynamic response obtained with the present D³ design and that obtained with the HOFCAS design. As mentioned above the bandpass with aircraft power matches that of Phase VI of the AFCAS program. The reduced response of the HOFCAS design is produced by the limited power available from the HPA. The limited HPA power reduces the power available to the torque motor coils resulting in a reduced loop gain under dynamic conditions. This produces reduced response but no decrease in static accuracy.

For comparison purposes the response curve of the ABU mode is shown in Figure 111. This response is the same as that of the two previous flight test programs and demonstrates that the modifications for HOFCAS did not affect the ABU operation.

RECOMMENDATIONS

The results of the laboratory tests demonstrate that the HOFCAS configuration is satisfactory for flight testing in a T-2C aircraft. The tests prove the HOFCAS concept is a practical approach for operating remotely located surface actuation with immunity to electromagnetic interference. The D³ actuator is ideally suited for this application because of the integrated actuator concept and the ease of adapting optical control the actuator.

The next logical step in developing the HOFCAS would be to demonstrate the concept in flight. Several additional developments, some of which are already in work, are needed to establish this concept for use in future high performance aircraft. These items are:

- A suitable optical actuator feedback link.
- High temperature digital electronics suitable for packaging inside the actuator.
- An improved rotary type valve-torque motor suitable for packaging inside the actuator.
- An improved HPA with the capability of handling the peak transient power requirements.
- Design of the integrated actuator package.

The optical feedback link and the valve-torque motor are already under development in separate Navy programs. The drive electronics would be the next needed development. The new electronics and the improved valve-torque motor can be expected to reduce the power requirements of the HPA.

Other studies of interest would include research into methods of improving the HPA efficiency without increased size, weight or complexity. A design study to show the advantages of the HPA over the "brute force" approach (shielding, filtering, etc.) to protect against EMP would also be desirable.

The conclusions of this study show that HOFCAS approach incorporating a D³ actuation system will produce a simple highly reliable actuation system that is immune to the effects of electromagnetic interference.

Figure 112 shows an actuator concept in which the digital electronics, rotary valve torque motor, and feedback transducers are integrated inside the actuator body.

INTERFACE OF FLIGHT CONTROL COMPUTER COMPLEX, D³ ACTUATION, AND HOFCAS

The HOFCAS could be configured for either three or four channel redundancy. With the improvements in digital technology and self monitoring techniques, it is anticipated that future systems will be triplex. A triplex redundancy concept for HOFCAS is illustrated in Figure 113.

The force motors will be designed with three windings per motor and will use a switching amplifier for each coil. This redundancy level will remove the need to disconnect a failed unit. Short circuit protection will be included at all six switching amplifiers. The amplifier is commanded with a PWM signal. The optical sensors are designed to supply a digital feedback signal to the computer. This system produces an actuation loop that is completely digitally controlled. The removal of interconnecting wires between the computer and the actuator will greatly reduce the susceptibility to electromagnetic interference. This system is still in the early stages of development and the redundancy concepts may be modified as additional test data is obtained on system components.

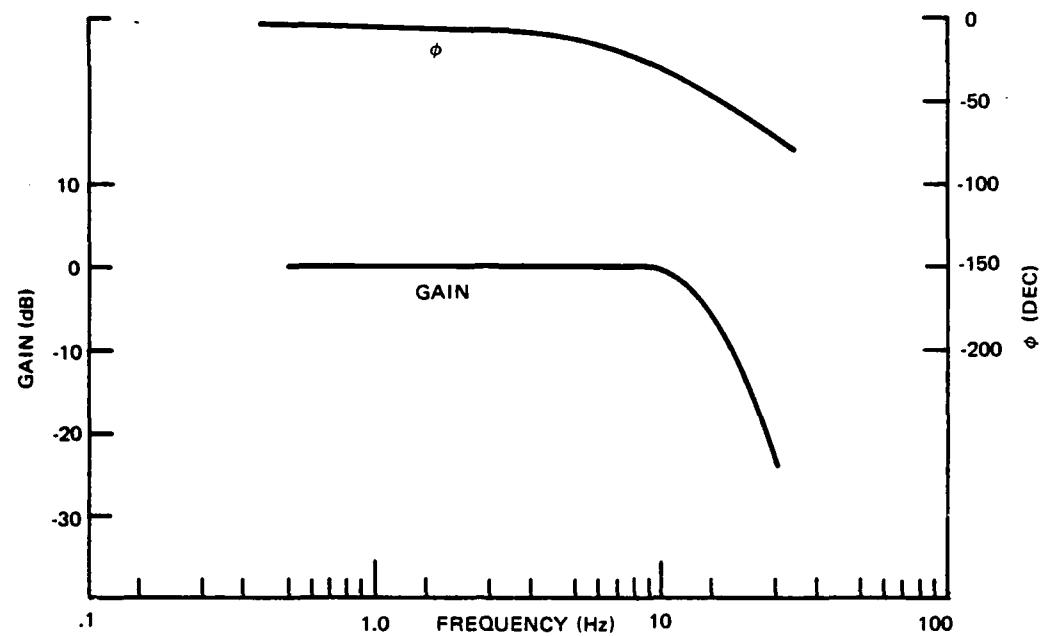


Figure 108. AFCAS Frequency Response

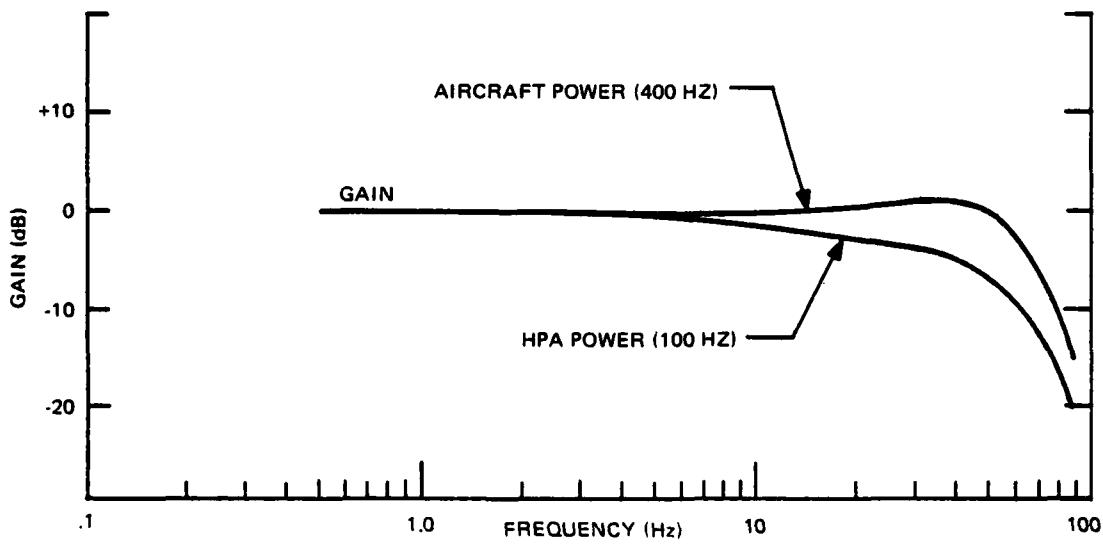


Figure 109. Digital Fly-By-Flight Mode System Response, Maximum Capability, HPA Power (1000 Hz) and Aircraft Power (400 Hz)

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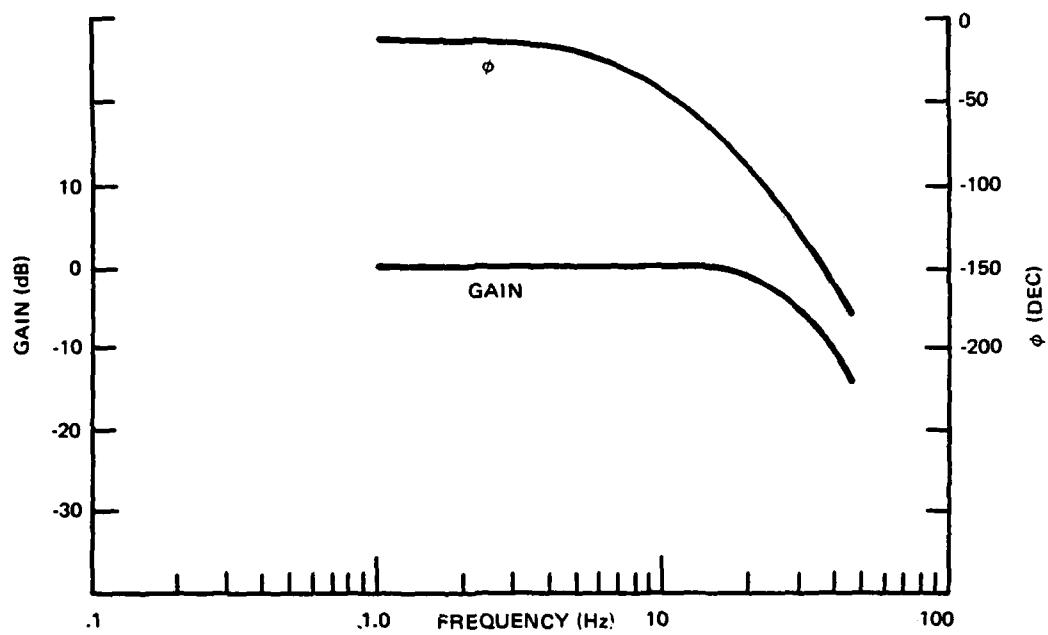


Figure 110. Frequency Response, AFCAS Designed for D^3

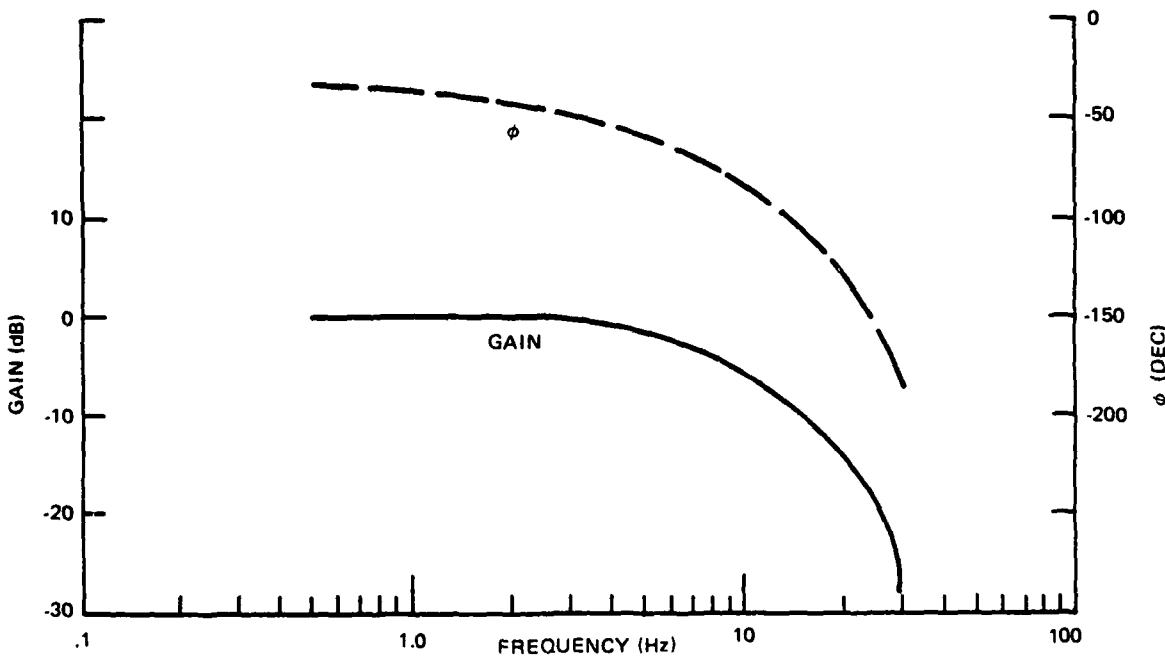


Figure 111. Analog Back-Up Mode System Response, Flight Gain Configuration, Aircraft Power (400 Hz)

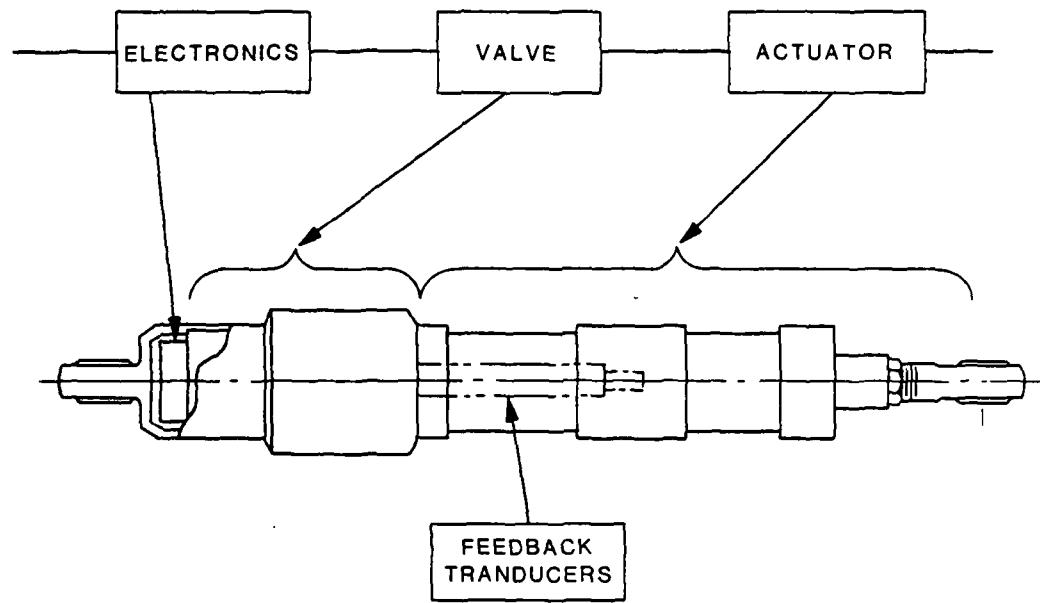


Figure 112. HOFCAS System Redundancy Concept

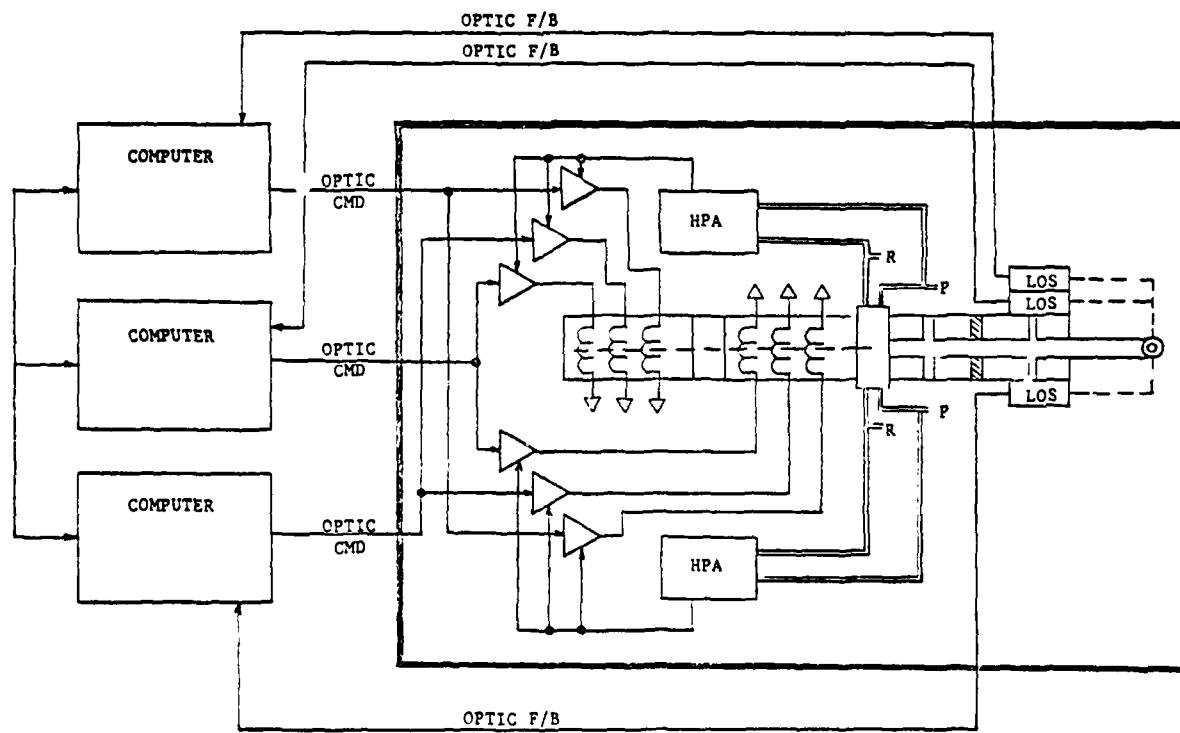


Figure 113. HOFCAS System Redundancy Concept

FAULT-TOLERANT ACTUATION CONCEPT FOR A RESEARCH TEST AIRCRAFT

BACKGROUND INFORMATION

Current hovering aircraft capabilities demand considerable visual contact flight before final landing, with the pilot providing most of the attitude-stabilizing, position-fixing, height-controlling and deck motion compensating functions. Pilot workload, even in clear weather operations, is excessive. This places additional demands of considerable magnitude upon the pilot and the flight control capabilities of the hovering vehicle. The development of an advanced, precise, and highly reliable flight control/guidance system concept to meet operational goals is a prime requirement. The manual and automatic modes of the flight control system must have sufficient authority to perform their required functions during the critical vertical take-off and landing operations. This implies that the conflict between automatic control authority and flight safety must be resolved by incorporating fail-safe and fault-tolerant features in the Flight Control System (FCS).

The study vehicle selected for the integration and evaluation of the 4-valve fault-tolerant actuation concept, developed by Bell Helicopter Textron Corporation, is a helicopter with a Type II power boosted FCS. The Type II FCS is a reversible control system wherein the pilot effort, which is exerted through a set of mechanical linkages, is at some point in these linkages boosted by a power source, which in this application is hydraulic. Flight control commands from the pilot and the Automatic Flight Control System (AFCS) are algebraically summed and then transmitted to the flight control surfaces.

Automatic landing capability is highly desired, which in turn dictates a large control power requirement. Therefore, the AFCS is anticipated to be a high (50 percent) authority system. During automatic landings on small ships, all ship kinematics, aircraft range, and range rate information will be transmitted to the helicopter via the Landing Guidance Sensor (LGS) data link. All computations of the flight control laws required for execution of the landing task will be performed by the Flight Control Computers (FCC) which will then transmit the appropriate commands to the flight control surfaces via the 4-valve fault tolerant AFCS actuator.

The actuation system for the AFCS is recognized as a critical technology in the development of a research flight test vehicle. It is for a dual fail-operate requirement that the 4-valve actuation system is designed.

FLIGHT CONTROL ACTUATOR REQUIREMENTS

In order to resolve the conflict between automatic control authority and flight safety the laboratory versions of the 4-valve AFCS and the primary boost actuators were required to have the following design features and capabilities:

- (a) The hydraulic and electrical power supplies shall be configured to provide a Failure Tolerance Level (FTL) of single-fail/operate for the primary actuators as well as for the AFCS series hydraulic actuators.

(b) The AFCS series actuators shall incorporate the following features:

- (1) Be dualized and include a feature that automatically centers and mechanically locks the piston shaft when the AFCS is disengaged or in the event all hydraulic or all electrical power is lost. The centering function shall center the series actuators at a rate that shall not overburden or disorient the pilot.
- (2) Be interfaced with the primary boost actuators in such a manner that full manual reversion capability is provided in the event all hydraulic power is lost.
- (3) Operate in conjunction with parallel trim actuators in such a manner that they shall normally be working about the center point of authority.
- (4) Be capable of receiving triplex electrical command signals from the AFCS and adding them algebraically to the pilot's mechanical input.
- (5) Possess at least a single-fail/operate capability with inherent fault-tolerant features.
- (6) Provide 50% of the total (stop to stop) command authority for the triplex electrical signals from the AFCS.
- (7) Contain provisions to disengage the triplex electrical signals. Disengagement will not effect a transient that could be considered hazardous.
- (8) Be powered by two independent 1,000 psi hydraulic systems.
- (9) Be installed so that triplex signals shall not be reflected back to the pilot's cockpit input device unless the summed input from the pilot and series actuator is in excess of the total allowable control travel.
- (10) Simultaneous application of large pilot AFCS electrical inputs shall not overload the output mechanical devices downstream of the actuator (stops downstream of the summed output). "Smart" constraints for structural protection should be considered.
- (11) Be efficiently packaged for installation in confined areas.

INTERFACE OF THE PRIMARY AND AFCS ACTUATORS

The actuator configurations shown in Figures 114 and 115 functionally describe two concepts on how the 4-valve AFCS actuator can be operationally interfaced with the primary boost actuators to effect a series-type control input.

The configuration shown in Figure 114 uses two grounded-base actuators with the two outputs differentially summed at a specific ratio to establish the desired AFCS authority. Stops are located downstream of the summing to limit total travel of the controlled output and hence to prevent overtravel of the basic controlled element. The AFCS actuator is dualized and uses 4-position feedback sensors, a centering spring/lock mechanism, and 4 electrohydraulic servo valves (EHSV) that are controlled by quad-redundant control paths. The quad-redundant electronics can be interfaced with digital or other analog systems.

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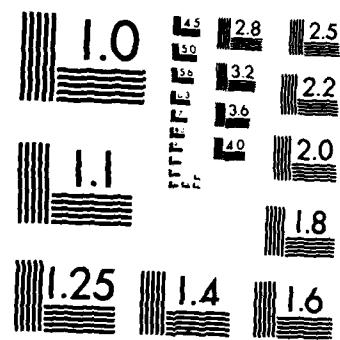
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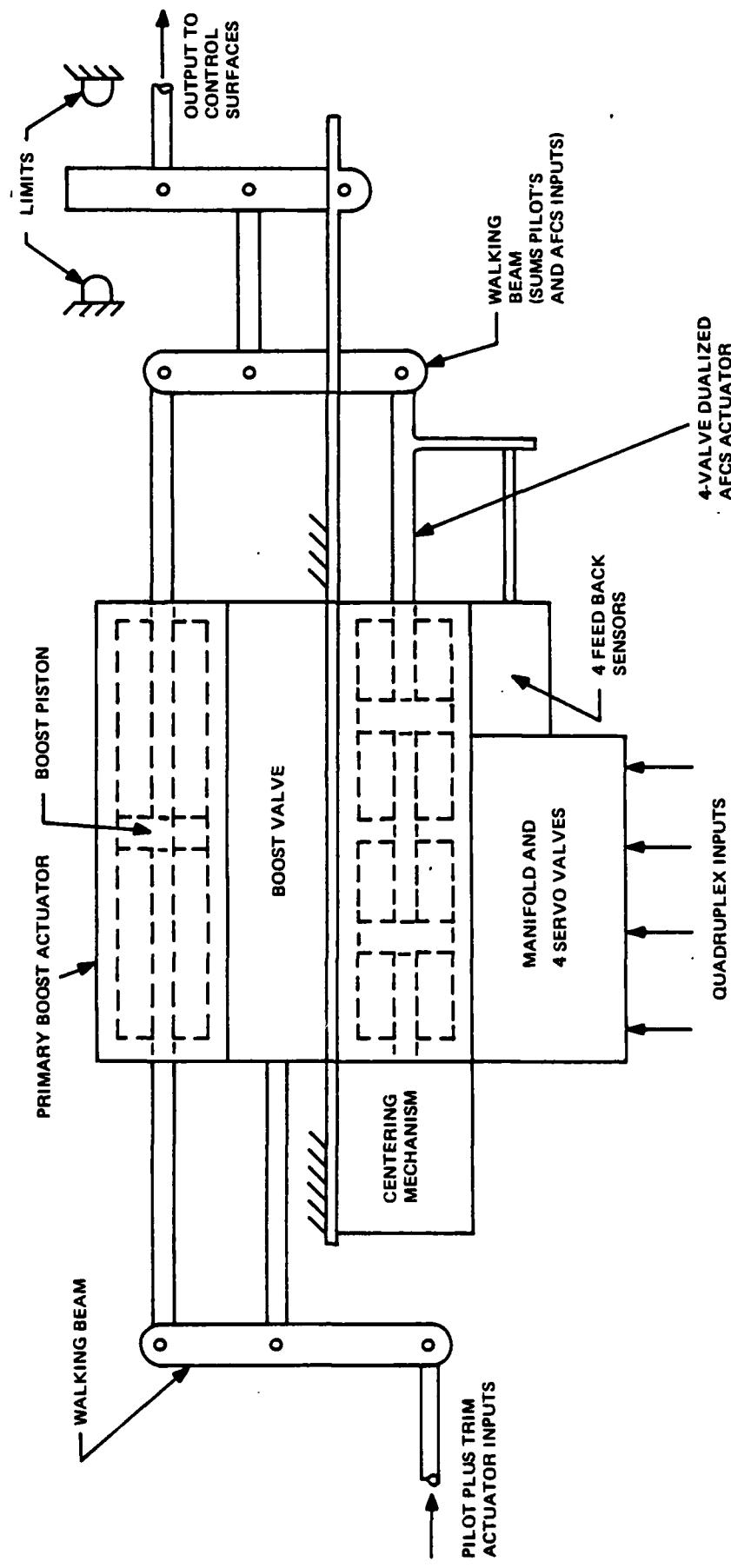


Figure 114. Output Summing Attached Actuators Configuration

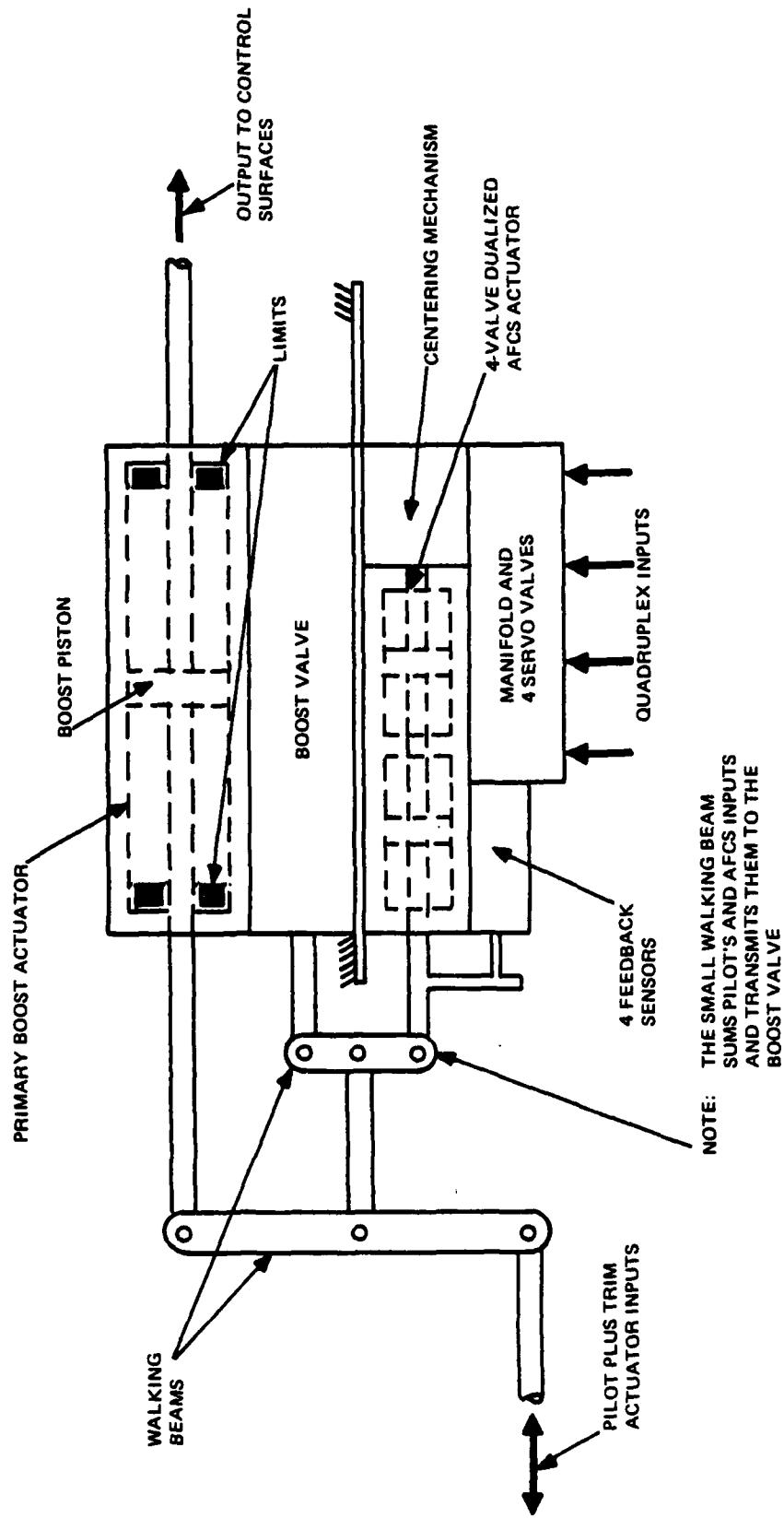


Figure 115. Input Summing Attached Actuators Configuration

Operationally the configuration shown in Figure 115 is the same as the Figure 114 configuration with the exception that the output of the AFCS actuator is mixed with the pilot's input to effect a differentially-summed output to the control valve for the primary actuator. This configuration uses the primary actuator for the muscle which allows the summing to be accomplished at a lower force level. Hence, a smaller actuator can be used. The centering spring/lock mechanism, however, will be required to have the force capability of centering in the event of a total loss of hydraulic power. Bell Helicopter Textron (BHT) has used an actuator configuration of this type in a 3-axis SCAS for the JetRanger helicopter.

This configuration was selected for laboratory because it offers a weight and size advantage as well as a comparatively low hydraulic flow requirement.

THE BELL HELICOPTER TEXTRON FOUR-VALVE, DIRECT DRIVE, ACTUATION CONCEPT

This actuation concept was originally developed by the BHT Corporation for use in 4-channel helicopter FBW control systems. It does not require the use of secondary actuators and its 4-valve configuration enables it to be highly fault tolerant. The 4-valve concept has high hinge moment and stiffness capabilities and can be used to actuate large swash plates and conventional flight control surfaces. It is readily adaptable for use in high performance fixed wing aircraft.

In this program for the development of a fault-tolerant actuation concept for a test aircraft the 4-valve concept will be used only for adding AFCS signals to those of the pilot's. It was chosen for this flight safety critical task because of its fault-tolerant features and high inherent reliability. Therefore, its force output requirement for this application is very low, being required only to actuate the boost valve of the primary boost actuator (see Figure 115).

1. Summarized Description of Concept

The basic fault-tolerant actuation system consists of dual hydraulic primary actuators, quadruplex electrical control paths, a failure management system, and electrical drive signals. Two electrical control paths are used for each piston. The failure management system is mechanically interfaced with the electrical control paths to provide maximum security. It provides automatic disengagement of a control path and also provides track error signals that are used in the control paths for automatic alignment of the four valves.

A flight test model of this system would include a master control panel and an annunciator panel. The control panel would provide the necessary control functions, preflight checkout capability, and a manual reset for each control channel. The annunciator panel would indicate the operating condition of the control paths and would operate in conjunction with the control panel for the pre-flight checkout.

2. FBW Control Paths

A control axis of the basic 4-valve actuation concept consists of four FBW control paths and a dual piston power cylinder. The four control paths connect the pilot's control input to the power actuator cylinder and include the four electrohydraulic servovalves as shown in Figure 116.

The dual actuator schematically depicted is conventional, except that the control head (spool valve assembly) has been replaced with the four electrohydraulic servovalves.

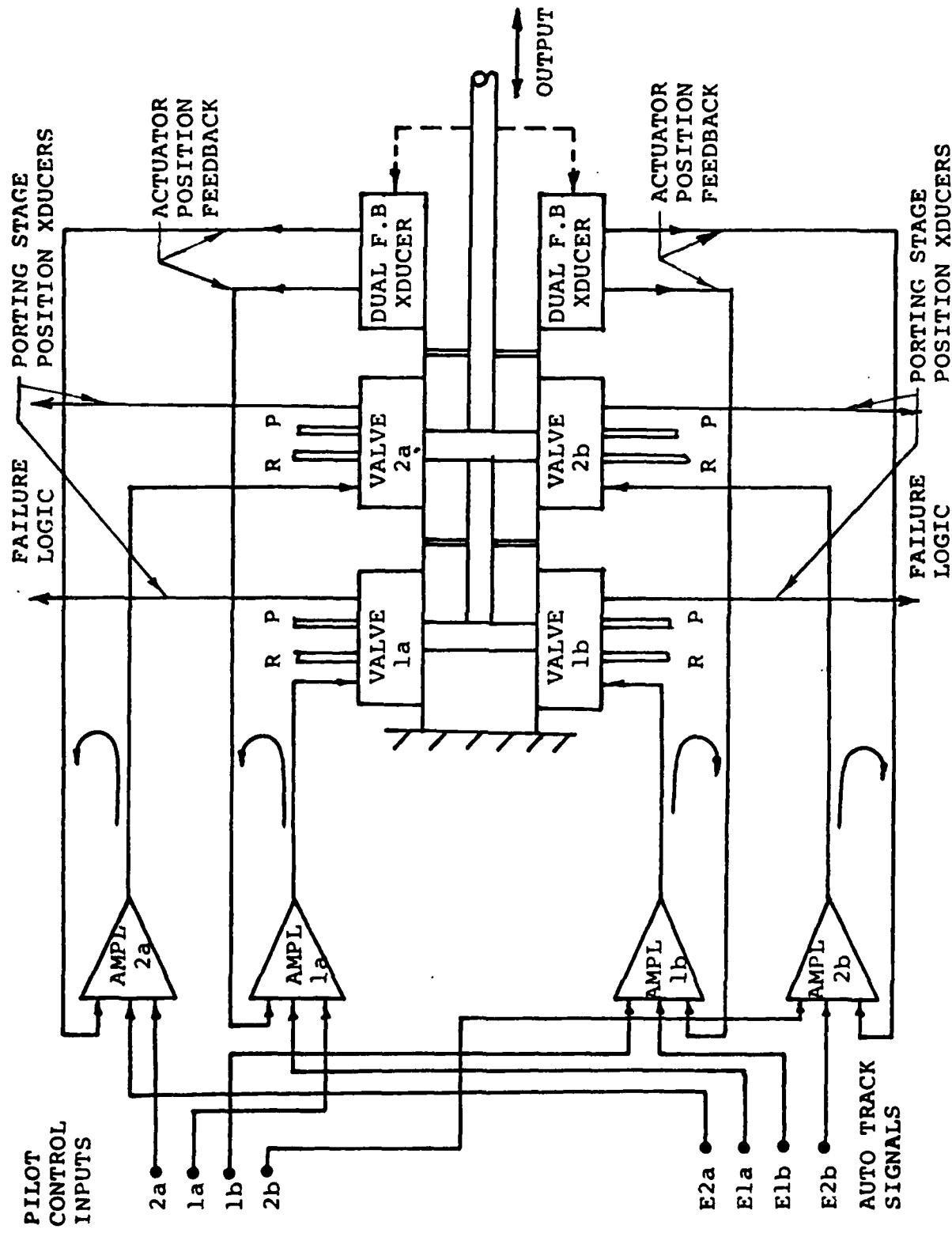


Figure 116. 4-Valve FBW Actuation Concept

Figure 116 is a functional schematic of the follow-up system; i.e., the dual actuator is slaved to the control inputs. All control paths are identical and operate simultaneously. A control input to the amplifiers proportionally opens the valves and drives the actuator until the dual feedback transducers provide feedback signals that cancel the command signals at the amplifier, which closes the valves and hence stops the actuator at a new position. The four valves are continuously and automatically aligned by a limited authority signal that is inherent in the failure management system (this feature is discussed in more detail later). The dual feedback transducers can be single elements and separately located to reduce vulnerability to battle damage if desired. The response of the actuator can be shaped to improve handling qualities as required.

The failure logic for the system shown in Figure 116 operates in the following manner. If a control path fails (e.g., path 1a), the path is automatically disengaged and Valve 1a is cut off to prevent leakage of fluid from one side of the piston to the other. A second path failure will be disengaged in the same manner. If the second failure should be path 1b, the logic circuitry will automatically engage a pressure-operated hydraulic bypass across the common piston so that the failure will not restrict the operation of the other piston. It is pointed out that if a first failure should disable the failure management system (described in the next section), the control path system, shown in Figure 116, has the inherent capability of absorbing a second failure. This is possible because, for example, if Valve 1a should fail and remain hardover, the other three valves will go in the opposite direction to oppose the actuator motion. This will effect a bypass around the piston common to Valves 1a and 1b and, hence, will allow the other piston to operate without any appreciable degradation. This inherent feature appreciably improves the overall reliability of the system and allows a comparatively simple failure management system to be used in place of a conventional voting scheme. The overall actuation system can be characterized as a forgiving type system.

3. Failure Management System

The failure management system is an integral part of the total system. It is required, as a minimum, to manage only one failure since the basic FBW system has an inherent failure tolerance level of single-fail/operate. Each control path has its own respective failure management unit which operates with a maximum of independence from the other failure management units. These units are mechanically interfaced with their respective control paths to provide operational security. In the event of a failure, the faulty control path is automatically disengaged. If two control paths that share a common piston fail, and hence are disengaged, the failure logic circuitry will effect a by-pass across the piston so that it will not adversely affect the operation of the other piston. A condition monitor is provided as a part of the failure management system to inform the pilot of the operational status of the FBW system. The monitor indicates soft failures as well as failed and disengaged control paths. In addition, the monitor provides a track error signal to each control path to provide continuous, automatic alignment of the servo valves which are the end elements of the paths.

The failure management system consists of a failure sensing function and an automatic detection disengage function. These functions are conceptually described in the following paragraphs.

a. Failure Sensing Function

The addition of an LVDT-type position transducer to the porting stage of the servovalves (second stage on conventional servovalves) allows the failure management circuitry to be mechanically interfaced with the control paths. This feature affords a more secure means of sensing a failed or degraded control path without reducing the reliability of the control path, and, hence, the transducer can be used to cover failures up to the power piston. Several other ways to provide a valve feedback signal for this failure monitor concept were considered. For example, differential pressure across the second stage of a conventional servovalve can be used. Also, the current flow through the first stage (flapper valve coil) can be used and is more economical. However, neither of these approaches

will provide 100 percent failure coverage and were discarded in favor of the valve position transducer approach. Valves of this type are currently available.

The basic failure sensing function for each power actuator channel is provided simply by using four equal value resistors in conjunction with the 4-valve position transducers. Connection of the resistors as shown in Figure 115 constitutes a very simple and reliable voting concept that allows each control path to comparatively monitor itself, determine a failure, and disengage itself.

Figure 117 is a simplified schematic of the basic concept for sensing a failure. For normal operation, the voltage across the valve position transducers should be the same. Since the voltages across the transducers are the same regardless of valve position, there will be no appreciable current in the resistors. Current will flow only in the resistors when the valve positions are not in agreement. If one control path has a "hard" failure, the respective porting stage will fully displace while the others will partially displace in the opposite direction. The voltage differences will cause a current in the resistor associated with the failed control path that is several times higher than the current in the other resistors, thus providing a means for identifying the failed path. For example, if Valve 2b is hardover, the other three will be displaced a small amount (depending on the actuator load) in the opposite direction and each will produce a transducer output voltage.

See References 20 and 21 for a more detailed description of the failure sensing function.

b. Automatic Detection/Disengage Function

This function is the part of the failure management system that detects the occurrence of a failure and isolates the fault by disengaging the affected control path. If the fault is not of sufficient magnitude to warrant a disengagement, it is presented to the pilot as a soft-fail (e.g., high null) indication. The soft-fail feature is a cautionary device for the pilot and constitutes a BITE function.

Several approaches for detecting failures were considered. One approach was to simply compare the magnitude of the failure voltage across each sense resistor with a set threshold. The second approach used a scheme for comparing the four failure voltages to determine a failure. This approach is not as simple as the first approach, but it appeared to be more tolerant to failures and was successfully used in the BHT 4-valve FBW feasibility program. Subsequent studies, however, indicated that an improved arrangement of the first approach has some advantages and, as stated above, was selected as the preferred approach. The major advantage of the first approach is that it has less failure modes and has a high degree of operational independence. The selected failure detection technique affords a failure management system that is very simple when compared to the more conventional voting schemes.

See References 20 and 21 for a more detailed description of the automatic detection/disengage function.

LABORATORY TESTS AND DEMONSTRATIONS

The objective of the laboratory test and demonstration program was to provide a means of appraising the 4-valve actuation system concept as a candidate actuation concept for use in the FCS of a potential test vehicle. The test program allowed the 4-valve actuation concept to be evaluated in terms of operational suitability audits ability to tolerate failures.

Tests were conducted to investigate the effects of the basic types of failures that can occur. The intent was to validate the 4-valve actuation concept as a viable fault-tolerant actuation system that

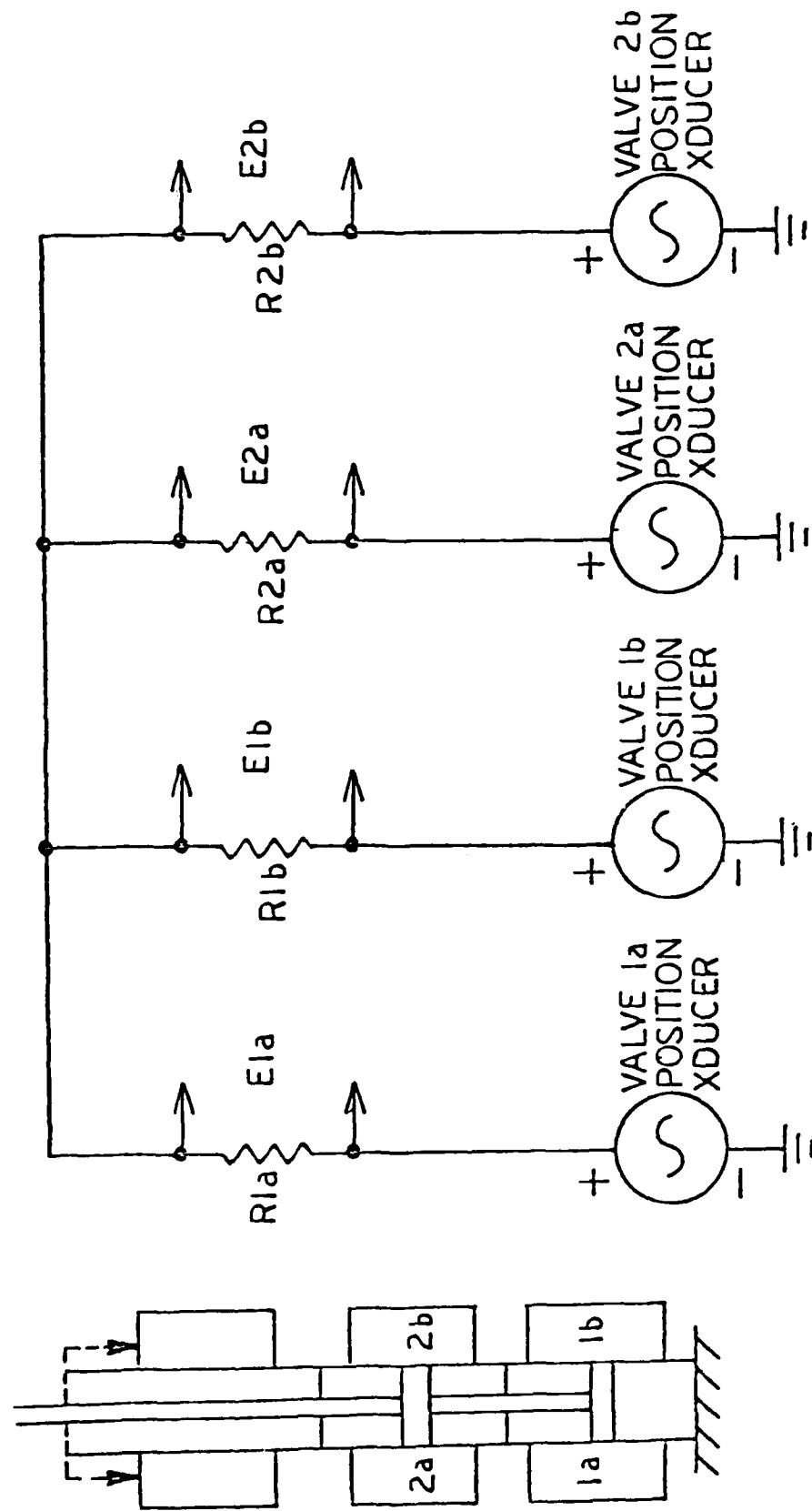


Figure 117. Fault Sensing Concept

can be interfaced with the FCS of the test aircraft. The AFCS control paths, up to and including the Electro-Hydraulic Servo Valves (EHSV's), were tested to assure a Fault Tolerance-Level (FTL) of dual fail-operate for the worst conditions. The electrical and hydraulic power systems were tested to assure that the failure effects on the total system would result in an FTL of single fail-operate and dual fail-safe. The failure modes were simulated using the switches on a failure simulation panel, which contained four electrical power switches, two hydraulic hand valves, and combinations of these input devices. Pertinent parameters were measured and recorded to define failure effects.

The test results demonstrated that the 4-valve actuation system concept is suitable for use in the FCS of the test aircraft. Tests validated the concept in terms of operational suitability and ability to tolerate failures.

The methodology used in the Integrated Test Program and the test results are fully described in Reference 20.

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